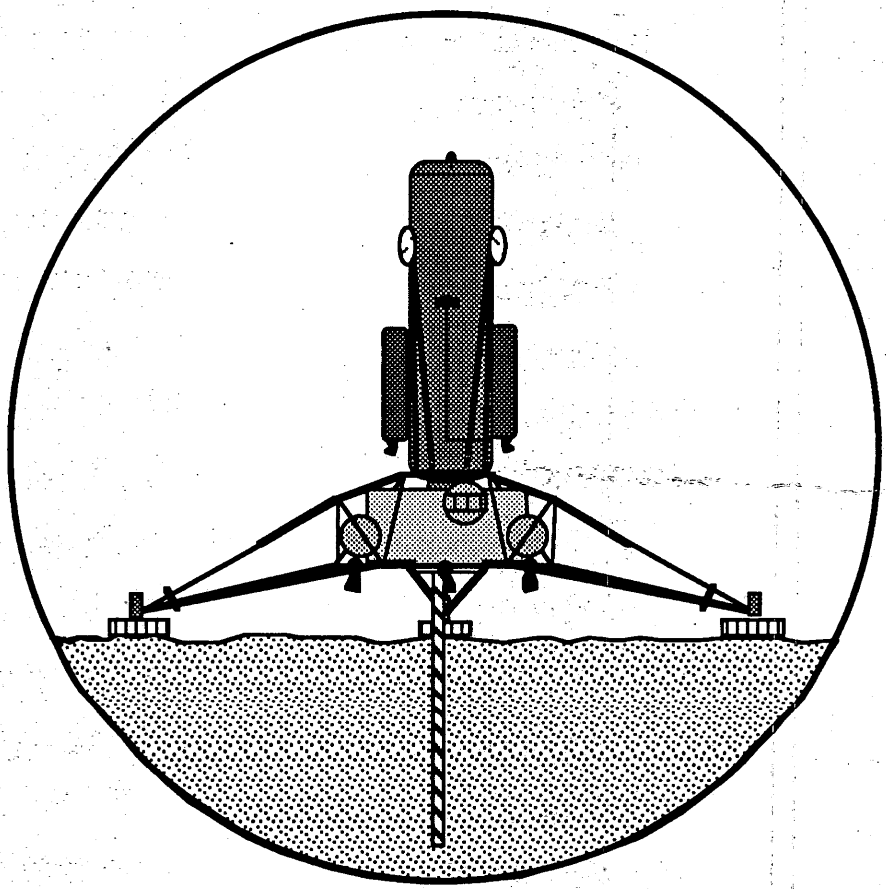
NASW-4435

# Oasis Lunar Systems

Presents a design study for a:

# Lunar Polar Coring Lander



Presented To:
Dr. Wallace Fowler
Department of Aerospace Engineering
The University of Texas at Austin

May 4, 1990

(NASA-CR-186684) LUNAR POLAR CORING LANDER Final Report (Texas Univ.) 158 p CSCL 03B

N90-25917

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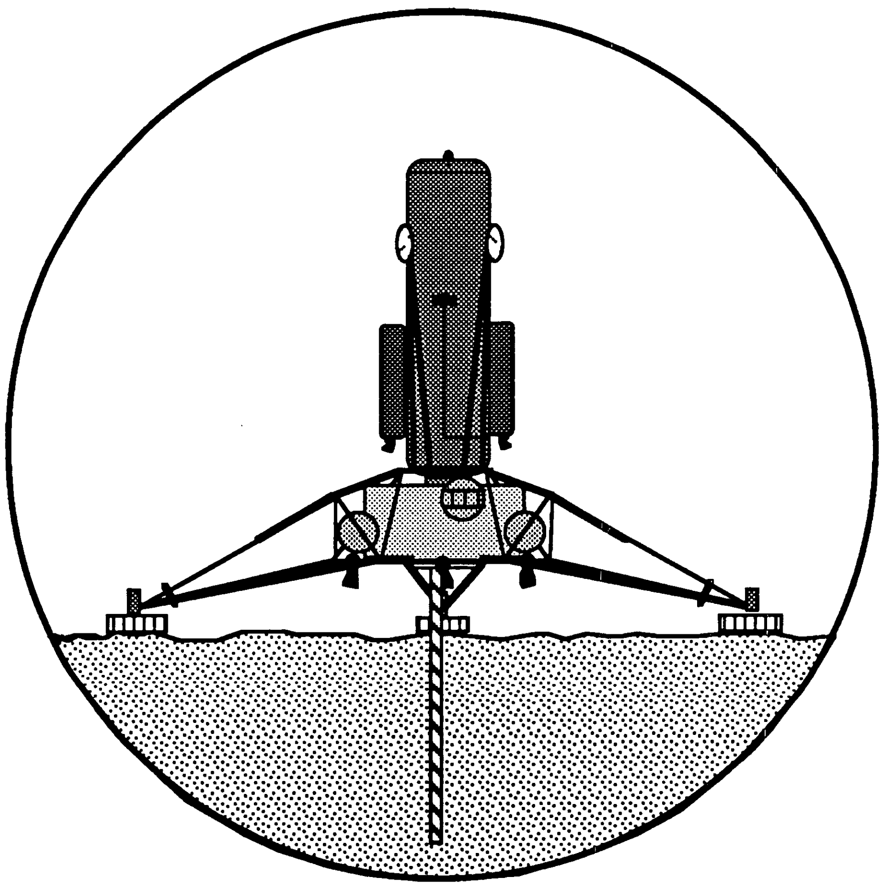

Presented To:
Dr. Wallace Fowler
Department of Aerospace Engineering
The University of Texas at Austin

May 4, 1990

## Final Report

# Lunar Polar Coring Lander

Submitted to:
Dr. Wallace Fowler
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May 7, 1990

## Executive Summary

As a new era in manned space exploration of the solar system begins, NASA is turning its sight back to the Moon. Plans to build a lunar base are presently being studied with a number of considerations. One of the most important considerations is qualifying and quantifying the presence of water on the moon. The existence of water on the Moon implies that future lunar settlements may be able to use this resource to produce things such as drinking water and rocket fuel. Due to the very high cost of transporting these materials to the Moon, in situ production could save billions of dollars in operating costs of the lunar base.

Scientists have suggested that the polar regions of the Moon may contain some amounts of water ice in the regolith. This report suggests six possible mission scenarios which would allow lunar polar soil samples to be collected for analysis. The options presented are: Remote sensing satellite, two unmanned robotic lunar coring missions (one is a sample return and one is a data return only), two combined manned and robotic polar coring missions, and one fully manned core retrieval mission. All the missions have their own advantages and all are considered to be viable with little to no required advancement of the present state of technology.

One of the combined manned and robotic missions has been singled out for detailed analysis. This mission proposes sending at least three unmanned robotic landers to the lunar pole to take core samples as deep as 15 meters. Upon successful completion of the coring operations, a manned mission would be sent to retrieve the samples and perform extensive experiments of the polar region.

Man's first step in returning to the Moon is recommended to investigate the issue of lunar polar water. The potential benefits of lunar water more than warrant sending either astronauts, robots or both to the Moon before any permanent facility is constructed.

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# 1.0 Introduction

Throughout the course of history, every time mankind has explored a new "world" or territory for colonization, the first priority has been to assess what natural resources were available in the new land. This has always been so and will always be. As mankind sets his sights on new-extraterrestrial-worlds, he must follow the same path.

The first step in the colonization of other worlds is the colonization of our own Moon. The first step in the colonization of our own Moon is the assessment of its resources available for use. Among the resources necessary for human colonization, water is one of the most important. Ever since the Apollo era, scientists have suggested that water may indeed exist at the lunar poles. The Apollo missions found no evidence of water in the equatorial regions; therefore, the next logical step in mankind's exploration of the Moon is to qualify the existence of water at the lunar poles.

This document presents a number of methods by which this objective could be accomplished. Six different missions are suggested and one of them is investigated in detail. This report is not intended to be a detailed mechanical design of a lunar vehicle, but merely a proof of concept investigation.

The designers of this project hope that the information presented will provide space scientist and engineers with some innovative ideas in approaching the topic of renewed lunar activity.

# 2.0 Lunar Environment

This mission will be targeting the lunar polar region as the most likely region to find water. This section will explain specifically why this region was targeted and will also explain some of the preliminary mission planning considerations for such a mission. First and most importantly, the issue of the existence of lunar water will be addressed.

## 2.1 Lunar Water

As was briefly mentioned earlier, scientists have theorized that certain concentrations of water may have remained trapped in the lunar polar regions. This section will outline some of the most prominent theories as to the sources of lunar water and some of the mechanisms by which it may have been lost from the lunar environment.

Watson, Murray and Brown (WMB) (1961) were the first to propose the possible existence of trapped water and other volatiles at the lunar poles. Their theories were first presented in 1961, since then many scientists have tried to verify their findings, mostly theoretically, but some through telescopic observations. Arnold (1977) has done a detailed analysis, based on the work of WMB, on the potential sources of lunar water and some destructive mechanisms. His work incorporates detailed photographs of the lunar poles taken during the Apollo era. There are four possible sources of lunar water which would have provided quantities of water which may be of practical interest, these are:

Solar Wind Reduction of Fe in the regolith
$H_2O$-Containing Meteoroids
Cometary Impact
Degassing of the interior

Two significant destructive methods are thought to exist: 1) photodissociation of $H_2O$ molecules from sunlight, 2) solar wind induced sputtering and decomposition of trapped $H_2O$.

The basic analysis in lunar water assessment is a thermodynamic one. Above a certain temperature (about 150-200°K) trapped water will sublimate to the lunar vacuum. Hence all energy sources must be investigated to assess their contribution to the total energy balance of a given region of soil. Two of the more important energy sources in this analysis are geothermal and solar wind. Reradiation from neighboring surfaces and lateral heat flow through the soil are two other inputs examined by Arnold. Arnold's analysis led to the conclusion that for a given region of shaded surface, the minimum linear dimension (of the shadow) necessary for trapping to occur is 30 meters. This is an important number as it identifies a minimum shadow size which can be targeted as a drilling site. A more detailed look at the sources of lunar water will now be given.

## 2.2 Reduction of Soil $Fe^{++}$ by Solar Wind Hydrogen Molecules

Theories suggest that lunar iron, which may have originated from meteors or regolith reduction, may have chemically reacted under the influence of solar hydrogen bombardment to form water as a byproduct.

The reactions would have occurred with ferrous silicates generally as,

$$Fe^{++} \text{ (in silicates)} + H_2 \text{ --> } Fe^{\circ} + H_2O.$$

Hapke et al. (1975) estimate that if all of the $H_2O$ were retained in the agglutinates, the mass fraction would be approximately 0.4%.

Langevin and Arnold (1977) suggest that in $2x10^9$ years, the lunar soil has been reworked and exposed to a depth of about 2 m. The amount of $H_2O$ corresponding to a 0.4% $Fe^{\circ}$ mass fraction would be 0.5 $g/cm^2$ .

## 2.3 Meteoric $H_2O$

Although scientists do not know for certain the amount of meteoric mass arriving at the Moon or Earth on a yearly basis, they have made estimates based on Earth watching satellite information (among other things). It is estimated that at least three tonne/day reaches the Earth and one tonne/day reach the Moon. It is believed that a large part of these objects are cometary in origin. The composition of these objects can be categorized into a number of groups, but the ones of interest to us are the carbonaceous chondritic and dense cometary material categories. This group of comets water content has been reasonably estimated to be approximately 3%. Although this number may seem high the extremely high velocity impacts will have most likely vaporized as well as dissociated a fair amount of the water content.

## 2.4 Cometary Impact

Relatively recent studies have found that about 250 objects with diameter greater than or equal to 1 m have struck the moon in the last $3.3 \times 10^9$ years. Comets are known to contain large quantities of water. Wetherhill (1976) suggests by analyzing the percentage of active short and long period comets and their probable impact velocities that in $2 \times 10^9$ years five comets worth of $H_2O$ has been put into the lunar atmosphere. He suggests that this would yield about $10^{16}$ to $10^{17}$ grams of $H_2O$.

## 2.5 Outgassing of $H_2O$ from the Lunar Interior

WMB had originally suggested that outgassing of water from within the moon may have been a possible source of lunar water. Most scientists presently do not regard this theory to be probable. However, the definite presence of gaseous $^{40}Ar$ has not allowed them to totally abandon this possibility. Let it suffice to say that the percentage of water emanating from this source would not be within any practical quantities.

## 2.6 Destructive Mechanisms

Two destructive mechanisms are predominant on the Moon. The first being solar wind and the second being micro-meteoroid impact (the gardening effect). These mechanisms serve to destroy the presence of molecular water.

This is done by either driving the water out of its trap where it is destroyed by solar radiation or by chemically dissociating the water molecule itself.

Arnold (1979) states that there are two potential destructive effects of solar wind: sputtering of whole molecules or groups of molecules, and chemical decomposition by radiation damage, with subsequent loss of hydrogen. These mechanisms are very prominent at the lower latitudes where the incoming solar radiation is nearly direct. At the higher latitudes however, the solar flux is significantly less which decreases the effectiveness of the destructive mechanism.

Meteoric impact, or gardening, is the process in which micrometeorite impacts "churn" the top layer of the lunar regolith. This happens due to the high velocity of the impact. Upon impact up to 10 times the mass of the meteoroid is ejected. In this process water is often vaporized, but is thought to also be churned to a lower depth or thrown to a nearby location. This destructive mechanism therefore is also a protective mechanism in that by burying water molecules it shields them from solar effect. Due to the nature of this process and the fact that it is not a frequent event, the amount of destruction caused by micrometeorite impact is believed to be minor.

A number of possible sources of lunar water have been presented and a number of methods by which lunar water may have been destroyed have been presented. The quantities of water are considered to be of practical interest. Considering only the cometary source of water; if it is distributed evenly through the polar trap zones, a concentration of about 5-50 $g/cm^2$ would result. If this material had been mixed through a depth of 2 m, as is suggested by Arnold, a mean concentration by weight would be on the order of 1-10%. This is certainly a quantity of practical interest.

This study has also eluded to a very important aspect of the proposed sampling mission, that is, how deep is the water. By combining the mechanisms described above, it has been shown by Arnold that the top 3-5 meters of lunar regolith would contain or would have retained the highest levels of water. Since outgassing has not been completely disproved and other unknown mechanisms may be present, it is suggested to search as deep as fifteen meters within a lunar pole crater to take into account all possible sources of water.

## 2.7 Lunar Topography and Landing Site Selection

An assessment of the potential location and abundance of lunar water has just shown that the regions on the Moon most likely to contain any quantities of water will be the large permanently shaded regions of the poles. Permanent shade with the amount of surface area required, will only be found inside the larger craters at the poles. This is because the very high incidence angle completely blocks out the sun at the depths within the craters.

A study was conducted of both of the lunar poles in order to understand as best as possible what type of geography will be encountered, as well as to give some preliminary landing site possibilities. Figure 1 through Figure 5 show the north and south pole. These photographs show that craters of the size of interest are indeed available and are actually abundant. The photographs also show that the south pole is significantly more mountainous than the north pole.

A number of variables will need to account for the landing site selections for this mission. First and most important, the data returned from a precursor remote sensing satellite mission, which is suggested for a complete topographical survey of the moon, will be crucial. The information provided from the precursor mission will be able to identify the chemical composition of the top strata of the the lunar regolith, thus identify most probable locations of trapped water. This information, along with the topographical details, will be most important for adequate landing site selections.

The photographs in Figures 1 through 5 show that the polar regions are very mountainous; hence, they are extremely difficult for any manned or unmanned lander to safely land in. Since there will be very little to no light at the lunar poles, visual contact will be very difficult, and mobility in a rover vehicle would be nearly impossible. These factors influence the mission planning as well as the spacecraft design. In terms of mission planning, suitable coring sites will be deemed as those sites which are permanently shaded and whose spectrographic analysis has shown that they contain relatively high levels of either water or hydrogen and oxygen. A limitation which has direct repercussions on the probe landing site will be the proximity of the manned landing site. The lunar pole has an interesting characteristic that the terminator does not stray very far over time. This is due to the fact that the moons inclination to the ecliptic is very small. The manned landing site will therefore be limited to landing on the light side of the terminator.

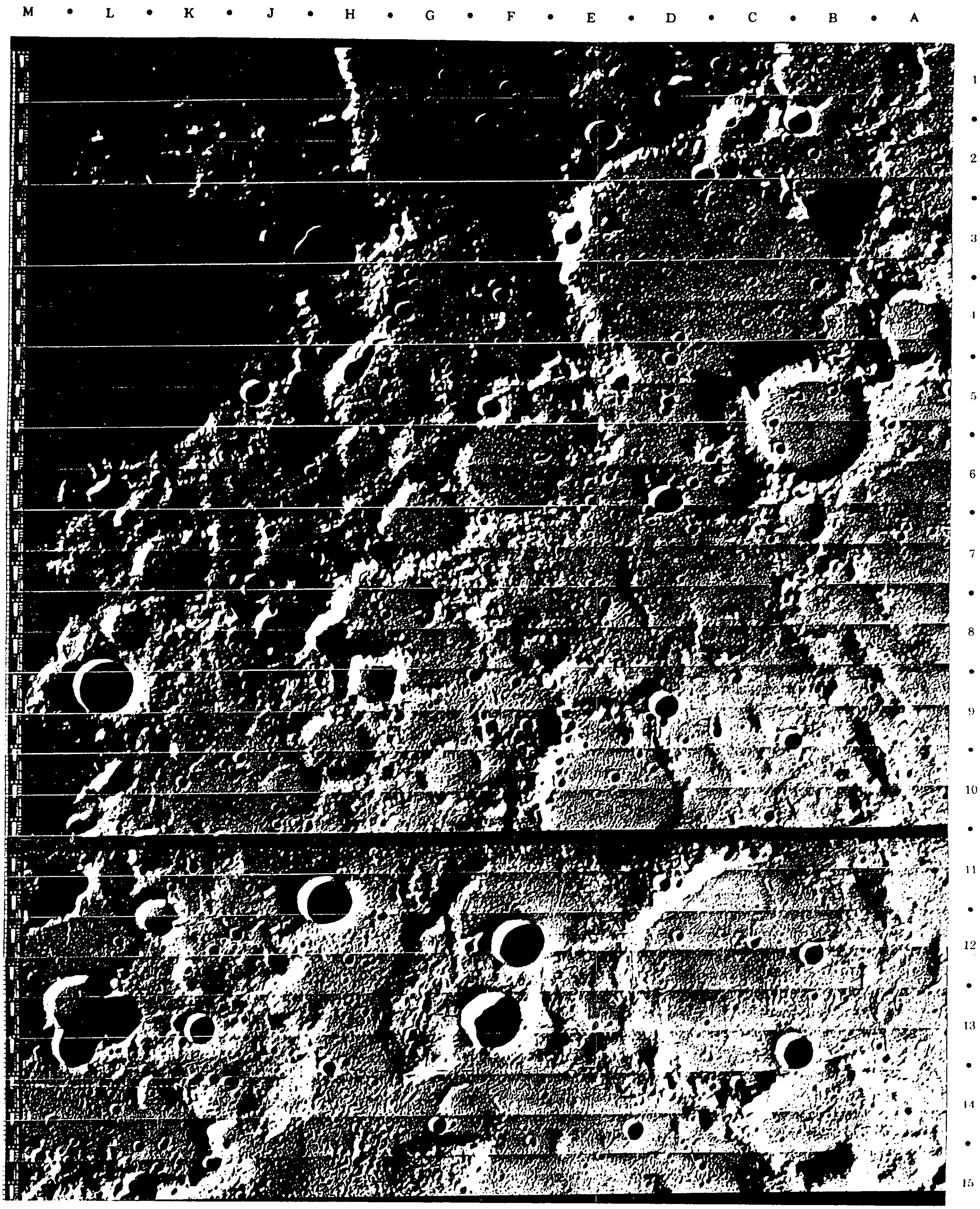


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PHOTO NO. IV-164-H3

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SUN ANGLE .......................... 77.1
SPACECRAFT ALTITUDE (KM) ...... 3345.81

N

1 CENTIMETER = 14 KILOMETERS

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MOUCHEZ ... C12

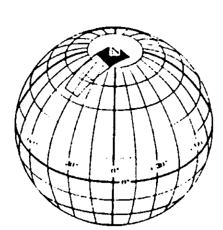

Figure 1: Lunar Pole map number one.

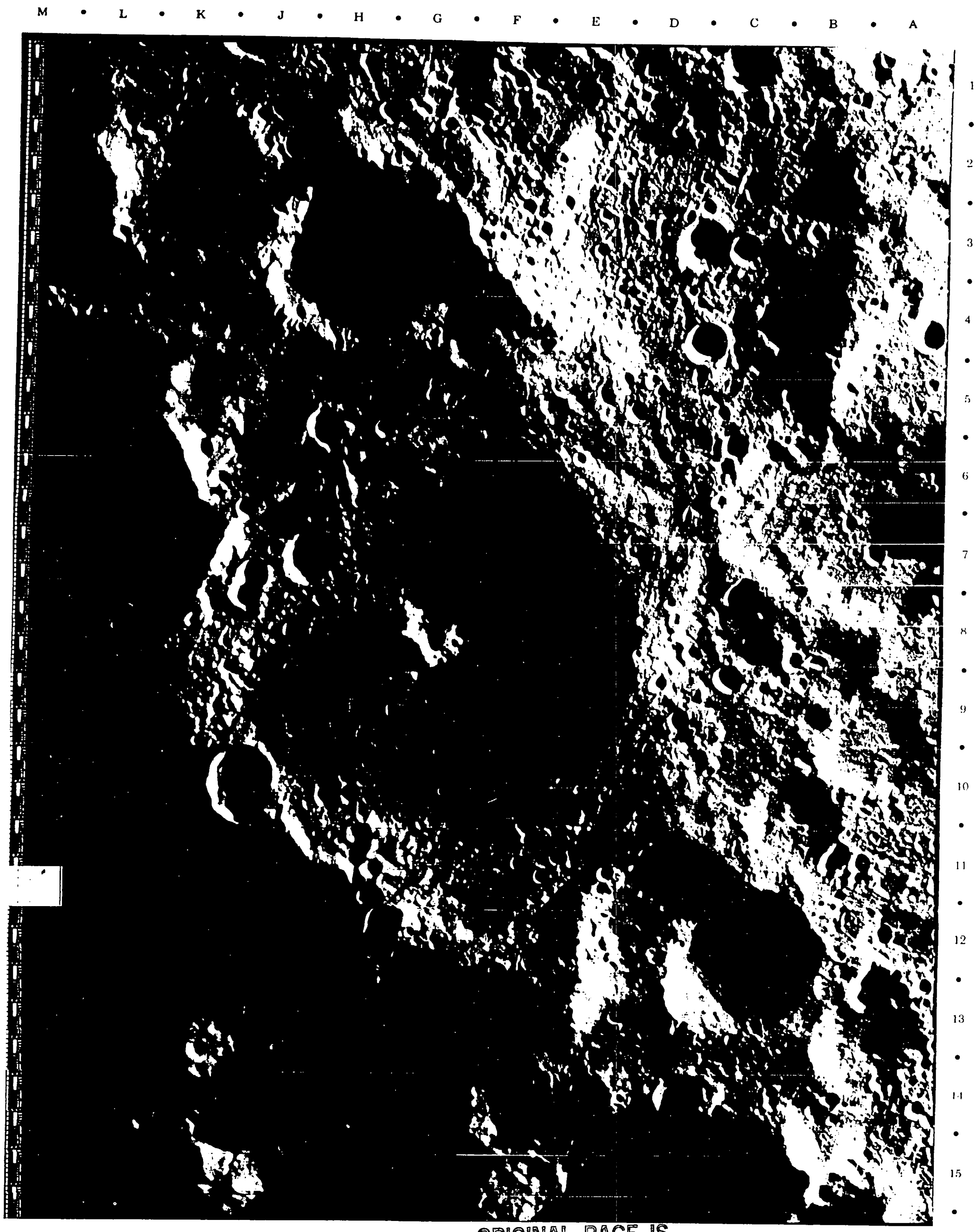




PHOTO NO. IV-193-H1

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SPACECRAFT ALTITUDE (KM) ..... 3519.21

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DRYGALSKI

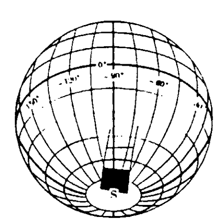

Figure 2: Lunar Pole map number two.

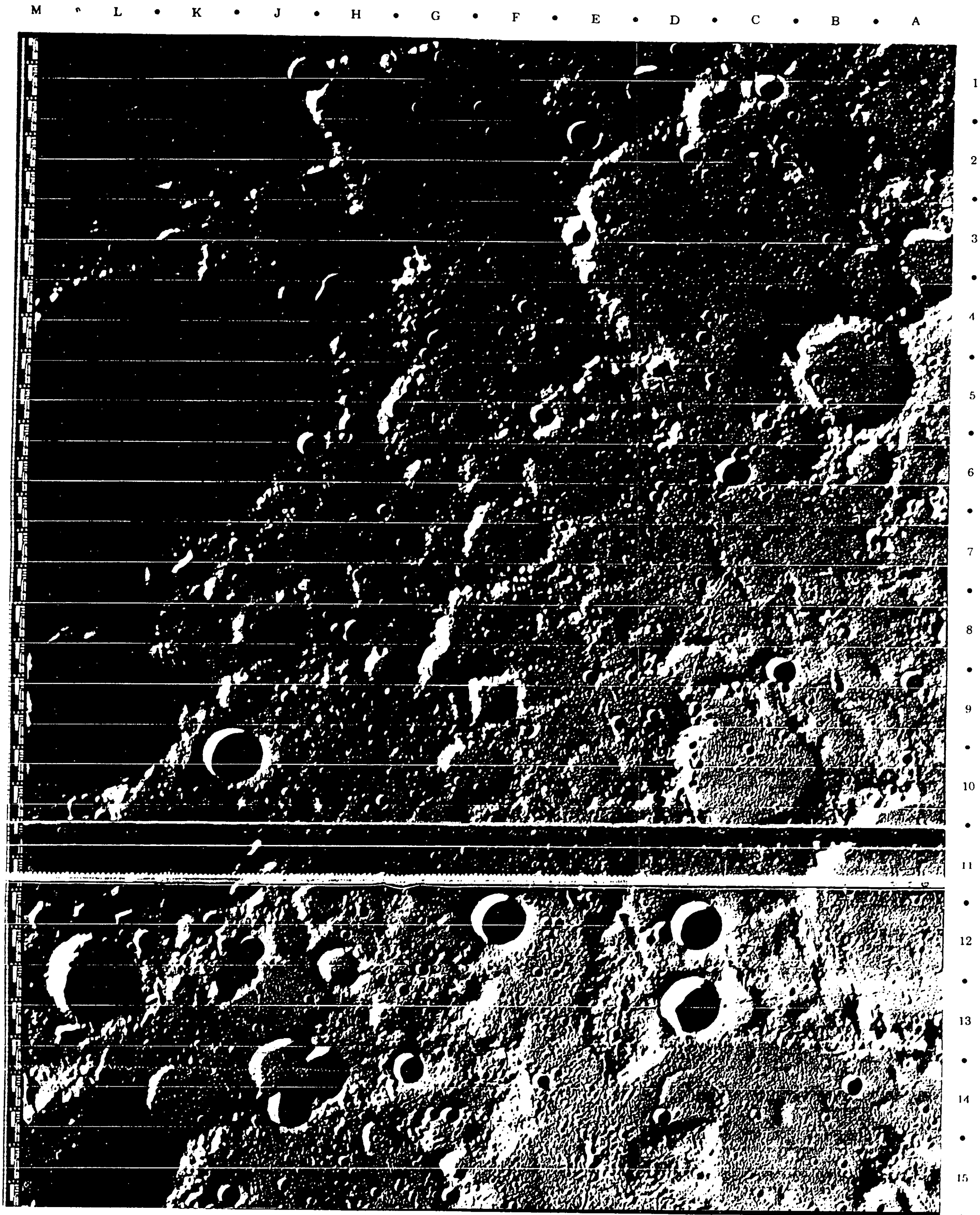




PHOTO NO. IV-176-H3
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SPACECRAFT ALTITUDE (KM)

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1 CENTIMETER ≈ 14 KILOMETERS

BYRD ......... D3
MOUCHEZ ...... A12

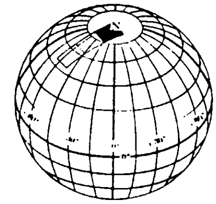

Figure 3: Lunar Pole map number three

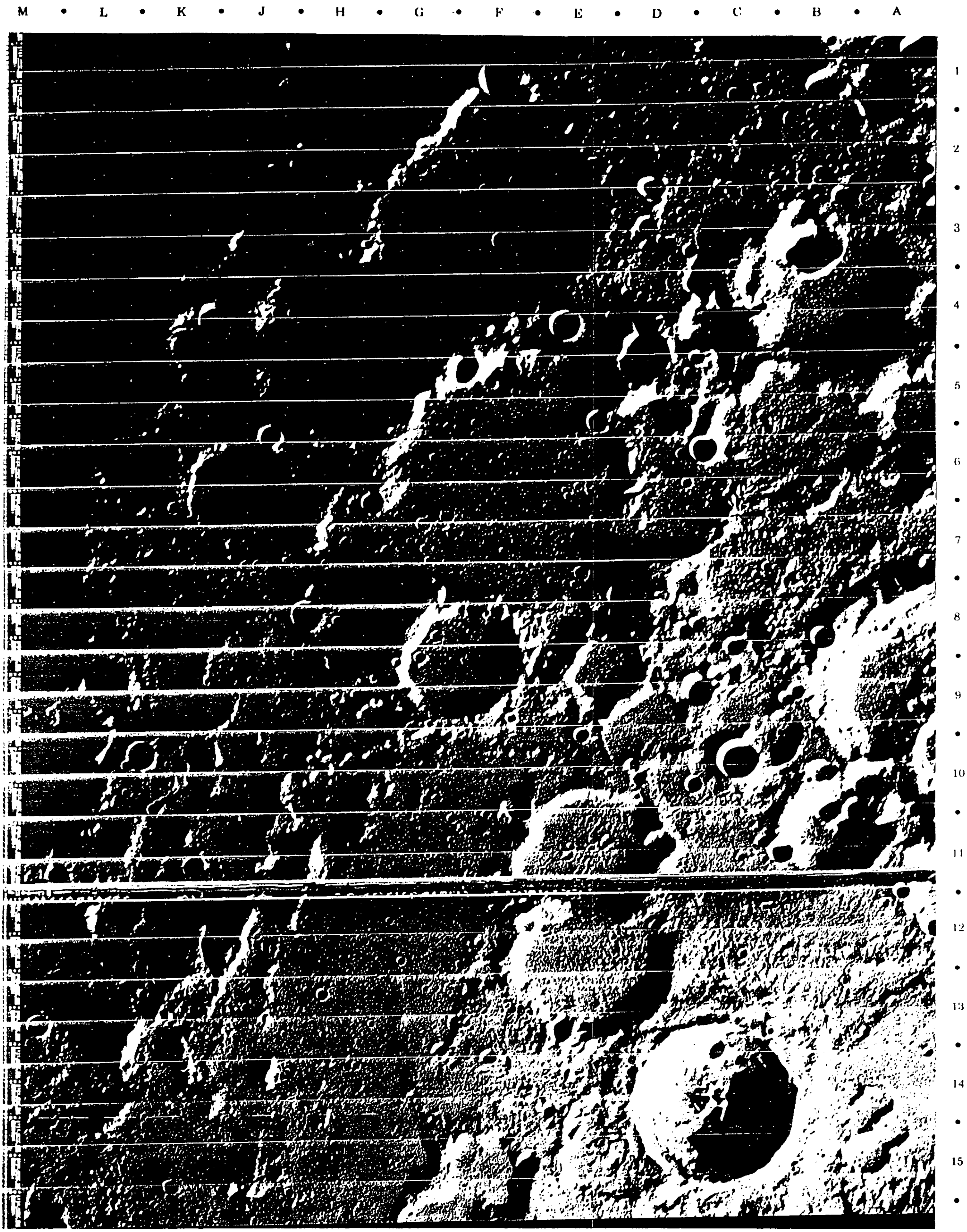


PHOTO NO. IV-116-H3

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N

1 CENTIMETER ≈ 14 KILOMETERS

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GIOJA .......... G9
MAIN .......... E11
PEARY ......... F3
SCORESBY .... C14

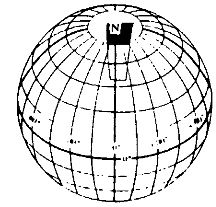

Figure 4: Lunar Pole map number four

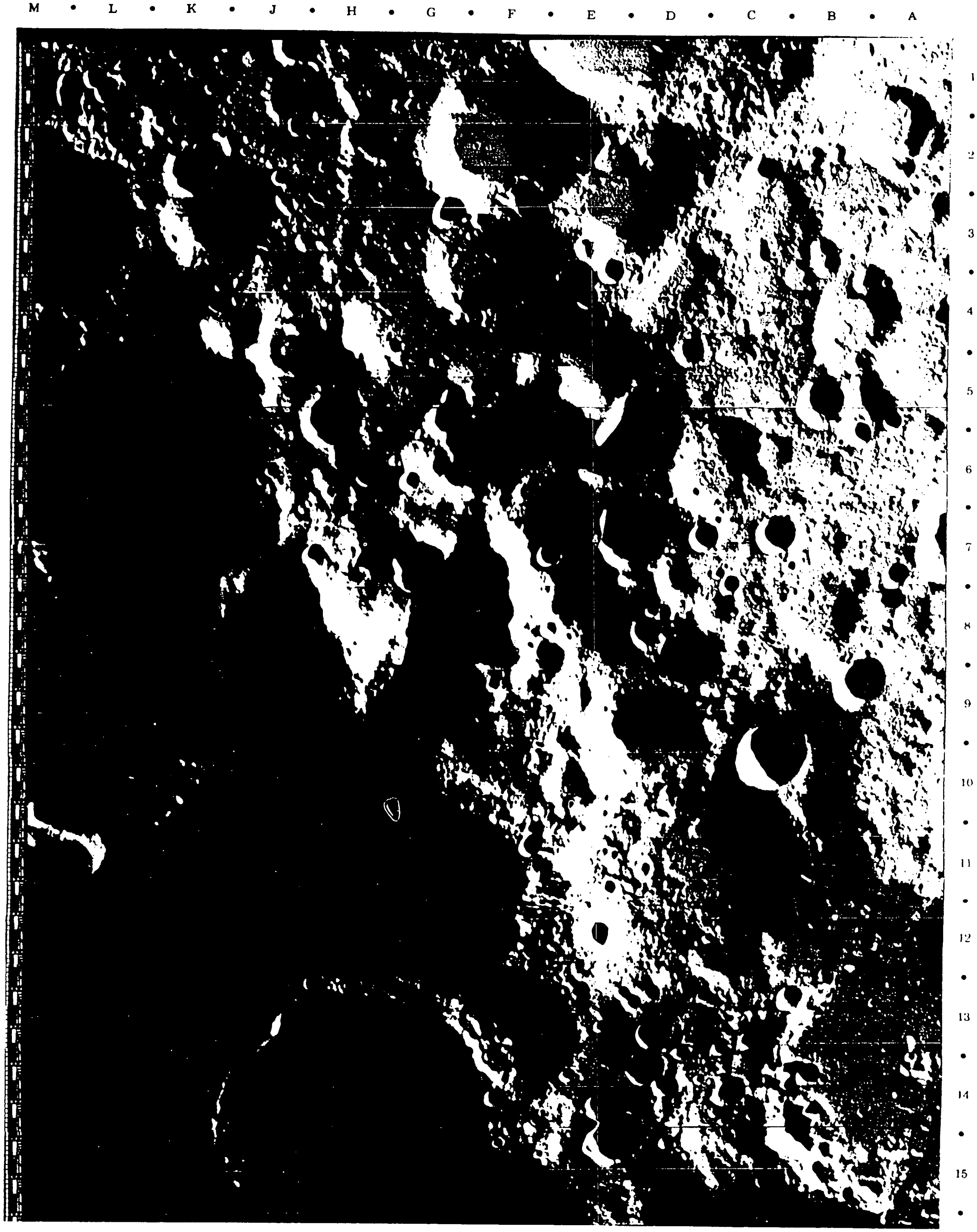


PHOTO NO. IV-154-H1

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SPACECRAFT ALTITUDE (KM) ..... 3613.43

N

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AMUNDSEN . H14

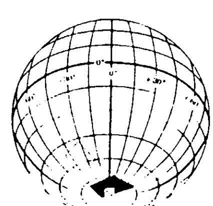

Figure 5: Lunar Pole map number five

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# 3.0 Mission / Vehicle Design

Due to the restrictive nature of the guidelines in the original RFP, greater latitude was desired in developing a viable set of solutions to the problem as perceived by Oasis. After consultation with the contract monitor, these guidelines were relaxed and modified to shift the focus of the design toward qualification and quantification of water at the lunar poles, thereby removing the absolute requirement that astronauts land to perform the drilling on the moon. These new guidelines led to the development of six possible mission scenarios which include a remote sensing satellite, two tele-robotic probes, two combined manned / robotic missions, and one fully manned mission. The following sections outline the design parameters of these missions and details what is achieved by each option. The mass requirements for the lander and the manned vehicles were modeled after the mass specifications of the Surveyor lander and the Apollo system respectively. The trajectory analysis and preliminary delta V calculations which set the guidelines for the mass requirements is presented in Appendix A. Computer models which were developed to give preliminary estimates of the propellent and structural masses required to place the landers on the lunar surface are listed in Appendix B. A description of the proposed mission scenario for each of the methods will now be presented.

## 3.1 Remote Sensing Satellite Mission

The first mission considered consists solely of launching a remote sensing satellite into lunar polar orbit. It is believed that such a satellite, equipped with a number of sophisticated imaging and spectral analysis sensing instruments, may be able to assess the existence of water on or below the surface of the moon (Lundberg, 1990). These instruments may include, but are not limited to:

1) Gamma-ray spectrometer,
2) Neutron spectrometer,
3) X-ray spectrometer,
4) Visual-near-IR spectrometer, and
5) 5-20 micron-IR spectrometer,
6) Multi-spectral imager,
7) Stereo imager, and
8) High resolution imager.
9) Synthetic aperture radar

Such an instrumentation package can obtain data on the lunar regolith pertaining to mineralogical, as well as elemental composition, and provide topographical imaging of the lunar surface (SMU,1986). From numbers provided on the mass of these systems, the sensing satellite is conservatively estimated to have a mass of 200 to 300 kg (SMU,1986).

Due to the absence of an atmosphere at the moon, the satellite will be able to orbit at a much lower altitude than any earth orbit. This lower orbit altitude and the absence of atmospheric interference will greatly enhance the quality of the scanning which can be done by the satellite. It is theorized that such a satellite may be able to scan up to a several meters below the surface of the moon. If the remote sensing satellite returned positive confirmation on the existence of water ice, the problem would then turn toward developing a method of excavation or extraction of the water. On the other hand, the satellite may only be able to determine that oxygen and hydrogen are present in proportions which indicate the possibility that water might exist in some form. The remaining design scenarios are based on the latter case, in which a sensing satellite found no conclusive evidence of water, but was able to determine the most likely sites at which to conduct a search. For this reason, it is suggested that the remote sensing satellite be sent as a precursor mission to any other missions. This will help determine mission requirements, provide mapping and a communications link between the lunar surface and the ground controllers on earth.

### 3.2 Fully Automated Data Return

This mission consists of sending a fully robotic drilling probe to the lunar poles to carry out in-situ soil analysis and relay the data to earth for examination by scientists and engineers. The lander corresponding to this mission is the lightest version of the probes with an estimated mass of 320 kg. This low mass allows several landers to be launched by a single rocket, thus maximizing the number of sites per unit cost at which the lunar soil can be analyzed. Assuming a three lander mission, a payload of an estimated 1500 kg will be placed in translunar orbit with 1000 kg landing on the lunar surface. Based upon these masses, the mission may be accomplished using only one Titan IV class launch vehicle.

The low mass of the probe also allows for the inclusion of a number of secondary scientific experiments which will be either attached to the lander or conducted in orbit. The types of experiments, however, and the information obtainable from them will be limited. Experiments must be designed to be robotically conducted and analyzed. The opportunity for hands-on analysis will not exist.

Since the analysis must be performed by the probe with the data relayed to Earth, and since the mission must be monitored from Earth, a substantial volume of information will be transmitted between the Earth and the moon. In order to facilitate this communication, an information relay satellite will have to be placed in orbit about the moon. This relay satellite can be the precursor remote sensing satellite.

## 3.3 Fully Automated Sample Return

The fully robotic sample return mission is similar to the Soviet Luna missions, in which soil samples are returned to earth for laboratory analysis. This mission entails sending an unmanned robotic lander to the lunar poles, taking core samples down to a 15 m depth, and launching the core samples back to earth.

In contrast to the data return mission, this mission requires the heaviest version of the probes with an estimated mass of 7600 kg. This large mass arises from the need to include fuel for the lunar re-orbit and transearth return as well as atmospheric reentry shielding. This substantial mass results in a limitation of one drilling probe per Titan IV class launch. This scenario will place an estimated 10700 kg in translunar orbit and an estimated 7600 kg on the lunar surface.

Additionally, the large mass of this probe limits the space available for the experimental packages. The data potential of this package is increased due to the retrieval and subsequent hands-on analysis of the experiments and core samples. However, the return of the core samples to the Earth poses another problem. To prevent any water content from being lost during atmospheric reentry, the sample must be hermetically sealed and possibly cryogenically cooled.

Although the core samples and desired portions of the experimental packages are returned to Earth, the transmission of information between the Earth and the moon will still be substantial. This mission, therefore, will require a satellite to be placed in lunar orbit just like the data return mission.

## 3.4 Manned and Robotic Lander Mission

The manned missions, just as the fully automated analysis missions, are based upon the deployment of several independently operated coring units. The major difference in this mission is that the core samples are retrieved directly at the lunar surface by a landing team. This requires that a manned mission be sent to the lunar surface in the style of the Apollo missions. This scenario also provides the opportunity to conduct an exceptionally large amount of scientific experiments on the lunar surface. The masses and equipment required by the manned portion of the mission are assumed to be approximately the same as those of Apollo. The robotic landers are sent to the moon prior to the manned mission and upon successful completion of the coring, the astronauts will be sent to retrieve the samples. After the astronauts land at the pole, they will select a sample retrieval landing site which will be a safe distance away from the LEM. The robotic landers will then launch the samples to the retrieval site in detachable containment units. When all the hoppers have landed, the astronauts will retrieve the samples. The astronauts will then complete all of the experiments before returning to Earth.

The robotic landers sent to the surface for this mission are relatively light because they will require only a small amount of propellent to perform the retrieval flight. The mass of a single lander delivered to the lunar surface has been estimated to be approximately 350 kg. By this estimation of mass delivered to the lunar surface, the mass of payload and propellent required to be put into LEO would be approximately 2000 kg for a single unit, and up to 6000 kg for three units. The mass of the individual landers is held to a minimum due to the fact that no secondary scientific packages need to be placed. The astronauts will perform all secondary experiments while they are on the lunar surface.

Using these mass estimates, up to three landers should be able to be launched from the earth using a Titan IV launch vehicle. Short of recommissioning a Saturn V, the most probable launch scenario for the manned missions would require multiple launches of payload and propellent and assembly in LEO. If a Shuttle C vehicle could be used, the number of launches needed to put the required mass into LEO for these missions could be cut considerably.

## 3.5 Manned and Robotic Orbital Rendezvous

The orbital retrieval mission uses methods very similar to the fully automated sample return mission but the heat shielding is deleted and the sample containment unit is greatly simplified because the core sealing requirements are less stringent. This scenario requires a manned lunar orbiting team but no manned landing. This provides an increased opportunity to gather scientific data relative to the fully automated units because of the direct presence of astronauts in a lunar orbit. The masses and equipment required by the manned portion of the mission are assumed to be approximately the same as those of Apollo; however, instead of a lunar landing vehicle, a core sample retrieval unit will be attached to the command module.

The drill units in this mission are sent prior to the astronauts, and when their drilling is completed, the manned vehicle is sent to lunar orbit to retrieve the core samples. The core sample containment section of the drill units is then commanded to launch out of its crater and match orbit with the manned capsule. The astronauts will rendezvous with the sample containers and extract the samples using a tele-operated retrieval system. Time will also be allocated to the manned mission to conduct scientific experiments before and/or after the rendezvous and retrieval.

The mass of the drillers used in this mission is relatively small, and may allow up to three drill units to be loaded onto a single launch vehicle. A single driller mass (as delivered to the lunar surface) has been estimated to be about 420 kg, and would require a total of about 2500 kg to be placed into LEO. This mass may allow up to three drill units to be launched on a single heavy launch vehicle, but it is more likely that a multi launch mission will be required as was suggested for the manned/robotic landing mission. These units have not been initially designed to carry any secondary scientific packages, but since astronauts will not be placed on the lunar surface, it may be desirable to integrate some scientific experiments into the lander units.

### 3.6 Fully Manned Landing Mission

This option is nearly analogous to the Apollo missions with the exception that the landings will be at the poles rather than the equator of the moon. The astronauts would use a lunar rover and portable coring unit to take core samples at a number of sites in the lander's vicinity. The samples would then be returned to Earth with the astronauts. The vehicle requirements are the same as those for the combined manned/robotic landing mission but the manned lander would be heavier since it would have to carry all of the drilling equipment plus the ECLSS needed for the extended period of the landing and drilling mission.

# 4.0 Mission Selection

Six mission scenarios have been offered as possible solutions. Each scenario has a unique set of operational characteristics, yet any one of them is capable of competing the prescribed task. Due to the overlapping characteristics of some of the missions, it would be impossible to choose one mission design based only on one or two outstanding characteristics without compromising other characteristics which may also be desirable. Therefore, a selection criteria matrix has been developed which provides an objective method of selecting and/or eliminating designs based on the desired characteristics.

The selection criteria matrix which is depicted as Figure 6, lists each mission across the top (as columns) and the characteristics along the sides (as rows). Immediately following the matrix is a description of each characteristic. A characteristic is depicted as a poor, weak, strong, or very strong asset of the mission by the presence of one, two, three, or four x's in the respective box. These characteristics are not the only criteria which are involved in the design selection, but they are representative of the major considerations of this process. Thus, by weighing the importance of these design performance characteristics on a relative scale, the least desirable mission(s) may be eliminated.

| | | | FULLY-AUTO DRILLER | | MANNED RETRIEVAL | |
|---|---|---|---|---|---|---|
| **MISSION SCENARIO** | SENSOR SATELLITE | MANNED DRILLING | ANALYSIS ONLY | SAMPLE RETURN | LUNAR LANDING | ORBIT RETRIEVAL |
| **SAFETY** | XXXX | X | XXXX | XXXX | XX | XXX |
| **RELIABILITY** | XX | XXXX | X | X | XXXX | XXX |
| **SECONDARY RETURN** | XXX | XXXX | XX | X | XXXX | XXX |
| **COVERAGE AREA** | XXXX | X | XXXX | XX | XXXX | XXXX |
| **INFRASTRUCTURE** | X | XXXX | XX | XX | XXX | XX |
| **COMPLEXITY** | XXXX | XX | XXX | XX | XX | XX |
| **COST** | XXX | X | XXX | XX | X | X |

Figure 6: MISSION DESIGN SELECTION MATRIX. This matrix represents the major design selection criteria for the proposed mission. A description of the characteristics listed is provided on the following page.

## 4.1 Characteristic Description

The characteristics used in the selection matrix are broad categories which cover a great deal of information, brief titles were given to the characteristics in order to fit them into matrix form. The descriptions below will clarify the meanings of these titles.

<u>Safety</u> What is the probability that lives or property might be lost?

<u>Reliability</u> This is a measure of how fail-safe the mission is. What is the possibility of failure, and can a mission gone awry be salvaged?

<u>Secondary Return</u> A measure of the capability to perform secondary scientific experiments concerning geological, mineralogical, gravitational, and other pertinent aspects of the lunar environment.

<u>Coverage Area</u> A measure of the diversity of sites which may be explored during a single mission.

<u>Infrastructure</u> Will the mission leave behind, or promote the development of any re-usable equipment or technology, or is it applicable only to this mission?

<u>Complexity</u> How easily can the mission be accomplished, and how much new equipment/technology is required?

<u>Cost</u> This includes not only the cost of the vehicle and its required subsystems, but also the cost of launching it from earth and the subsequent operational support, plus any retrieval of equipment,crew, samples, and/or data.

## 4.2 Analyses And Applications of Selection Criterion

Each of the proposed mission scenarios provides a viable method of determining the presence of water on the moon. The problem which arises at this juncture is the determination of which factors will be weighed most heavily in selecting a scenario. The following discussion on the importance of the mission characteristics is based on the interpretation by Oasis personnel of the original request for proposal.

### 4.2.1 Criterion Analyses

The most important characteristics of any mission are the safety and reliability of the equipment and mission operational procedures. Despite the importance of any other features of the mission, the weighting factors of safety and reliability must remain in the highest categories. The manned missions are unavoidably the most dangerous type of mission and will require a significant amount of redundant system capabilities in their designs, thereby driving up the overall system cost

Cost and complexity of an operation may seem excessively high for manned missions. Although these may be very important factors, the return which is derived from these missions may often override them. For this reason, cost and complexity are therefore relegated to slightly lower weighting categories.

Secondary return, coverage area, and infrastructure comprise the remainder of the characteristics in the matrix, and are very important considerations in the formulation of a continuous program of research and development of a lunar support system. In order to maximize the validity of the primary function of this mission, coring must be performed at diverse sites. This goal requires either a single unit of exceptional mobility, or a number of independent stationary units. The mass, power, and volume requirements dictated by the equipment which is designed to accomplish this primary goal will have a direct impact on the secondary return capabilities of the mission. This impact may be such that a mission which provides exceptional coverage may be of very little value (unless water is found) due to limited capability to conduct secondary functions. Missions of this type have another weakness in that such singularity of purpose contributes very little toward development of an infrastructure for future use.

### 4.2.2 Application to Mission Selection / Elimination

In considering the interpretation of criterion analysis derived in the previous section, application to the individual designs allows a comparison of the designs on a common scale. Therefore, the weakest of the designs can easily be pinpointed and eliminated.

The first design to be eliminated is the manual drilling mission. This scenario provides very limited coverage and is inherently the most dangerous due to the extensive time astronauts must spend on the lunar surface. Further elimination of designs however, is not so obvious. The safety of the unmanned missions appears to be a very desirable characteristic, but a synergistic consolidation of the secondary return and reliability characteristics of the manned missions may prove important enough to override this factor.

Another consideration which has not been presented is the fact that the remote sensing satellite may not provide conclusive evidence that water is present at the lunar surface. The information obtained by the sensing satellite can disclose the location at which the proportion of elements indicates the highest probability that water may be present in some form. For this reason, the sensing satellite should be sent as a precursor mission to any of the missions, either to locate a definite source of water, or to define the most likely sites at which to search for water.

Through this analysis of the six proposed design solutions, the fully manned operation has been eliminated due to expense, safety, and lack of coverage, and the sensing satellite has been relegated to a precursor mission. The design of the follow-on mission to a precursor satellite mission will be dependent upon the conclusiveness of the data which is returned by the satellite.

# 5.0 Design Commonality

Oasis has been presented with the problem of qualifying the existence of water on the moon. Each of the aforementioned mission scenarios achieves the primary objective as well as a unique set of secondary objectives. Differentiating between the scenarios requires a full knowledge of the secondary objectives. The remainder of this report, therefore, focuses on the one common denominator of all scenarios: the robotic lander. By focusing on this central element, Oasis leaves each of the scenarios as a viable option.

The robotic lander described hereafter is an inherently modular design, easily adapted to any of the scenarios with the addition or subtraction of certain components. Figure 7 illustrates the design commonality between the different mission scenarios. The probe essentially consists of three modules: drill core canister module, landing module, and drilling module. The drilling module consists of the mechanism for driving the drill into the lunar soil and is identical for all mission scenarios. Likewise, the landing module, consisting of the flexible tripod landing platform and the hover rockets, is identical for all scenarios with the exception of the totally robotic-sample return option. This scenario requires a larger, heavier payload and a landing module modified accordingly. The major difference in the various versions of the drilling probe are in the drill core canister module. This module consists of the cannister containing the coring equipment and has the following features for each of the mission scenarios.

Fully Robotic Sample Return

1. Rockets for returning drill core canister to Earth.
2. Atmospheric re-entry package with shield.

Manned Orbit Retrieval

1. Rockets sized to launch cannister into LLO.
2. In orbit rendezvous package.

Manned-Landing Retrieval

1. Rockets sized to launch cannister out of craters to collection site.
2. Landing structure for landing at collection sites.

Totally Robotic Data Return

Core sample analysis package.

In addition to the differences in the main modules, some differences may arise in the accessories, e.g. communication package, necessary for various mission scenarios.

For the purpose of an in depth probe design, Oasis has chosen the probe corresponding to the manned landing retrieval mission. Oasis has chosen this scenario for a number of reasons. First, this scenario best complies with the requirements presented to Oasis in the initial problem statement, i.e. manned mission with sample return. Second, Oasis considers this scenario the best and most feasible for the aforementioned reason. Third, this version of the probe is the most generic and regardless of which mission is ultimately chosen, it can easily be modified for use in that mission.

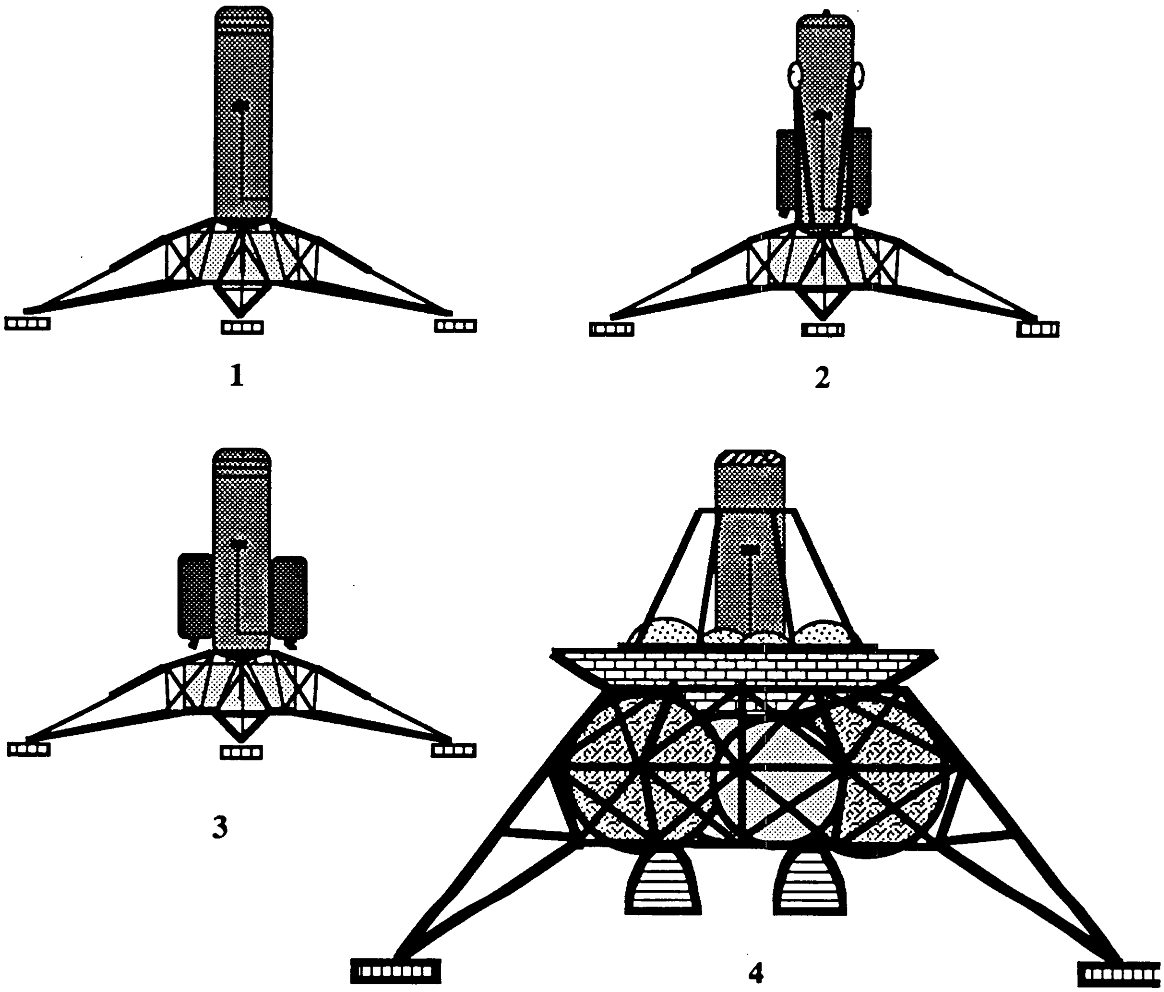
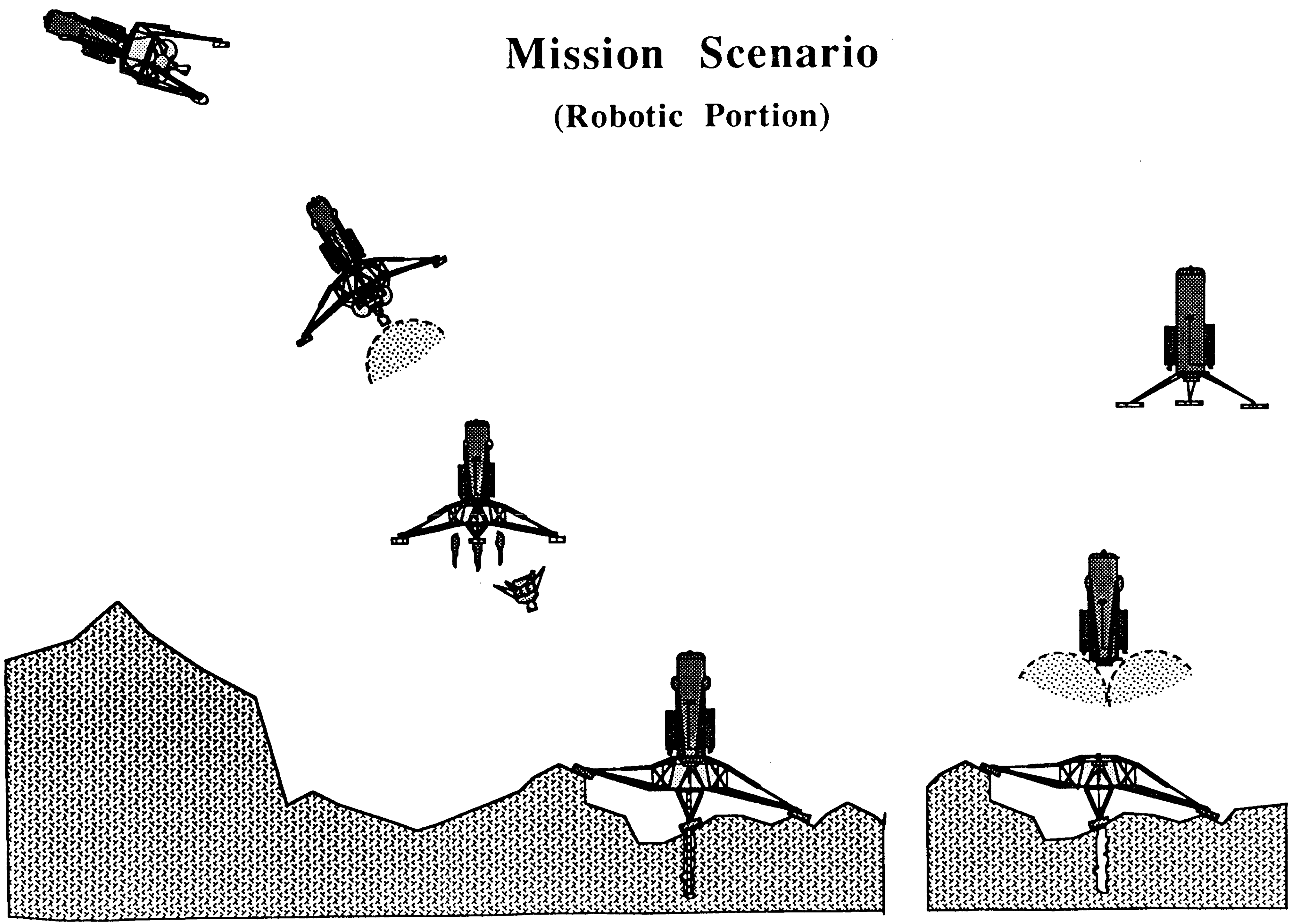



Figure 7: Commonality of the four robotic landers. 1.Data return mission, 2. Manned retrieval mission, 3. Orbital retrieval mission, 4. Totally robotic sample return mission.

# 6.0 Spacecraft Design

Figure 8 shows the mission scenario for which the spacecraft design will be based. Figures 9 and 10 show the final design of the lander and its nomenclature. A synthesis of unique subsystems is required for the implementation of any spacecraft design. The following sections describe in detail the subsystems which have been investigated.

Figure 8: Robotic lander mission scenario.



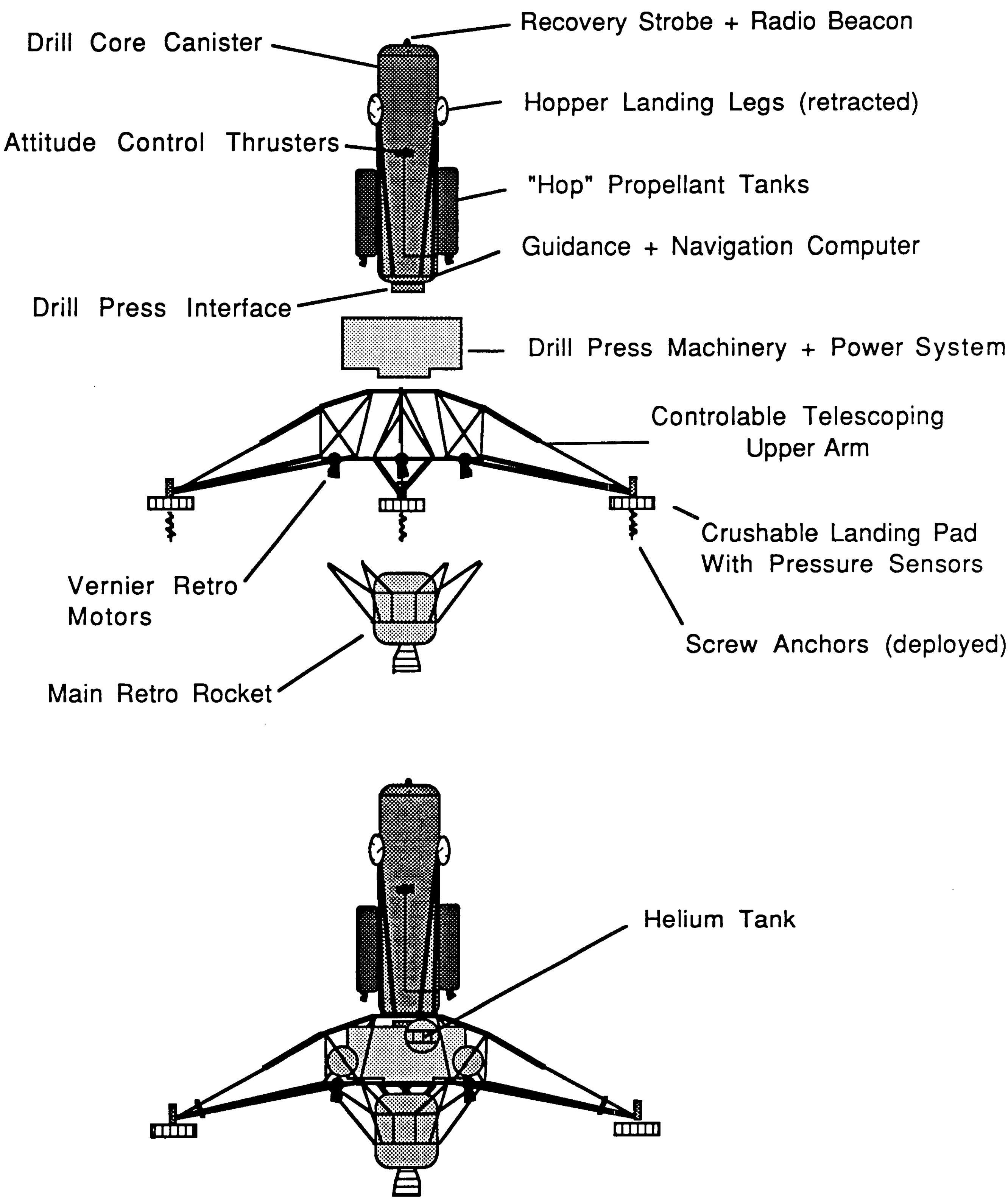
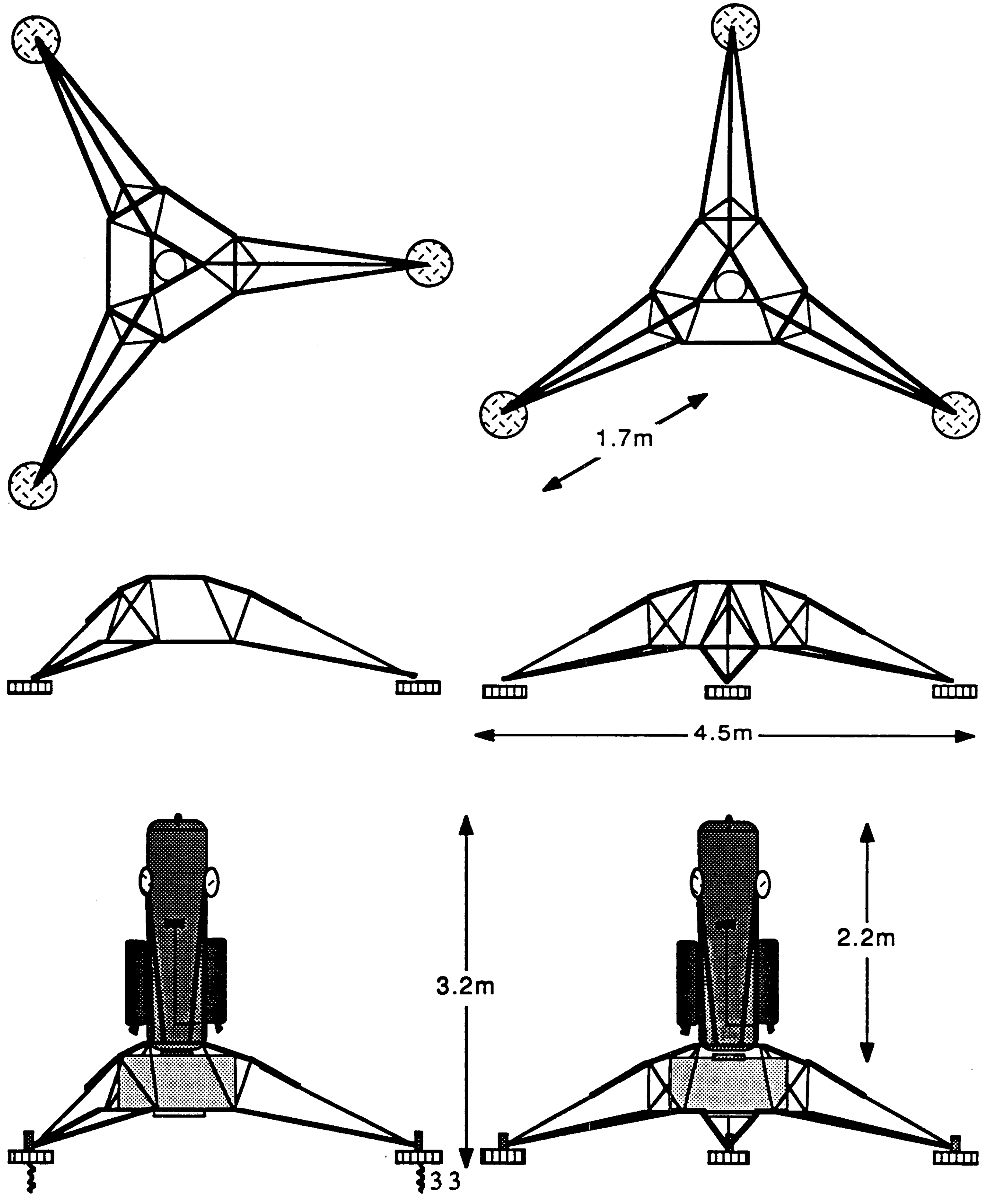



Figure 9: Spacecraft nomenclature and vehicle breakdown.

Figure 10: Lander frame and main systems integration.



## 6.1 Propulsion

The propulsion subsystem is divided into four main categories: main retro, vernier retro, attitude and guidance control and sample re-launch. Figure 11 shows the lander configuration with all of the propulsion modules.

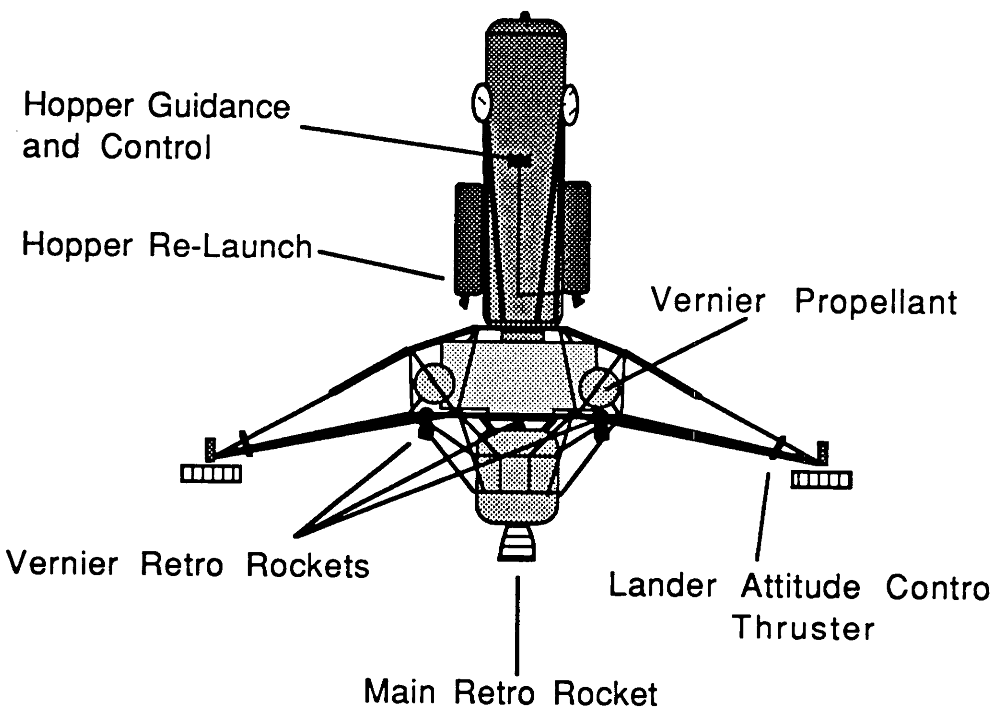



Figure 11: Lander propulsion system breakdown.

### 6.1.1 Main Retro-Propulsion System

The main retro-propulsion system's only function is to deorbit the lander by applying a delta V in the direction opposite to its orbital velocity vector. The deorbiting burn sequence will also be responsible for controlling the vertical velocity of the lander; hence, the burn will be a controlled vectored burn. In selecting a propulsion system, two options were considered: solid and liquid propellants. Solid rocket motors have the advantage of simplicity and reliability; however, they are not throttleable nor are they restartable. Liquid rocket motors have the disadvantage of higher complexity and slightly less reliability. They are also throttleable as well as restartable. The specific impulse of liquid rockets is generally much higher than solid rockets; however, a survey of applicable existing liquid rocket motors revealed that their average specific impulse was nearly the same as that of the solid rocket motors in the size class needed for the lander. Because of this similarity, it was decided to use a solid rocket motor for the retro-propulsion unit. Again, the drivers for this choice were the high reliability and simplicity of design.

Using preliminary lander mass and delta V requirements, the necessary propellent mass and motor/tankage mass were estimated. The resulting numbers are listed below. From these numbers it was determined that the main retro motor could be an off-loaded STAR-30 series motor, or an up-rated STAR-5 or STAR-17 motor. These motors provide a specific impulse in the range of 284 to 290 seconds and thrust in a range from 2000 to 10000 pounds.

| | | |
|---|---|---|
| Mass of lander (wet) | 340 | Kg |
| Main Retro Delta V | 1704 | m/s |
| Propellant Mass (main retro only) | 286 | Kg |
| Mass to be Deorbited | 626 | Kg |

An ejectable underside mounted propulsion unit was designed for the lander. By placing the retro rocket underneath the center of mass of the lander, the tremendous forces associated with the deorbit burn are more efficiently transfered to the lander and no undesired moments will be produced. This results in a much lower demand on the attitude control thrusters, thereby reducing propellent requirements. The retro rocket is mounted directly to the frame of the lander and is jettisoned after burn-out by detonating small pyrotechnic devices placed in the retro rocket support frame. These charges will break the attaching members and push the expended propulsion module away from the descent trajectory of the lander.

### 6.1.2 Vernier Retro Propulsion

The vernier retro propulsion system is responsible for stabilizing and controlling the lander after the main retro rockets are jettisoned. The vernier rockets will control the lander's vertical velocity as well as provide the majority of the thrust needed to do any cross range maneuvering. Due to the necessity for fine thrust control, a multi-start, storeable-liquid propulsion system was selected. This system consists of a trio of 300 pounds thrust, Rocketdyne RS-21 motors. Some of the preliminary sizing numbers are shown below.

| | | |
|---|---|---|
| Mass of lander (dry) | 306 | kg |
| Vernier Total Delta V | 220 | m/s |
| Propellant Mass (vernier retro only) | 34 | kg |
| Fuel Volume | .015 | $m^3$ |
| Oxidizer Volume | .014 | $m^3$ |
| Mass to Landing (wet) | 340 | kg |

The numbers shown above were calculated assuming that the main retro propulsive burn had removed 98% of the landers orbital velocity at burn out (10,000 m). The vernier motors must therefore provide sufficient delta V to to remove any residual velocity plus all of the landers velocity gained through an equivalent freefall height of 10,000 meters ( $V= (2gh)^{1/2}$ ). Sufficient propellent was also added to allow the lander to touch down at a near hover velocity. Although the lander will not actually hover, a small amount of fuel will be needed to maintain a lander touch down velocity of approximately 1/2 m/s. This will be accomplished by allocating enough fuel for 30 seconds of hover.

Three gimballed rockets are used in the propulsion system. Each rocket motor is mounted to the frame of the lander at the base of the landing legs. Gimballing is provided to allow the lander a small degree of cross range mobility without employing additional thrusters.

### 6.1.3 Attitude and Guidance Control

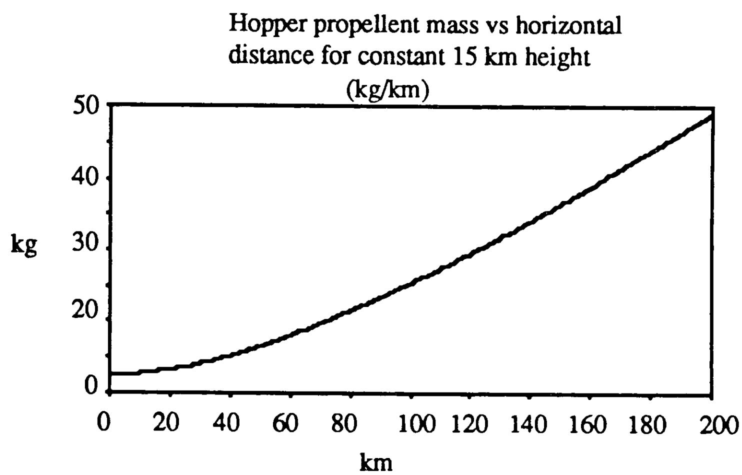
In order to be able to control and stabilize the probe during landing a simple attitude and guidance control system was designed. To maximize the efficiency of the system it was decided to place the thrusters as far away from the center of gravity as possible. This effectively ensured the largest possible moment arm which decreased the amount of thrust needed to yield a given amount of control. This system is very similar to the one which was successfully employed by the Surveyor lander. A small thruster is mounted near the end of each of the landing legs. These thrusters are responsible for controlling the direction of the main retro thrust vector and they are crucial in the touch down sequence. In the touch down sequence the vernier rockets will be responsible for controlling the descent velocity, but it will be the attitude control motors which will stabilize the lander and keep it vertical when (and if) its legs touch down on a rough, or non-level, site. The amount of fuel consumed by this system is very small compared to the vernier system. The attitude and guidance control motors will therefore feed off of the vernier propellant tanks. An approximately 5% increase in vernier fuel will be sufficient to fuel this system.

### 6.1.4 Hopper Re-Launch

The sample containment unit, the hopper, will require its own propulsion system in order to launch the samples to a designated landing area for pick up. Each hopper will have a range of 150 km, which provides a large coverage area and gives the astronauts more flexibility in choosing a desirable landing site for their Lunar Excursion Module (LEM). Two types of propellants were again investigated with three primary constraints: storeability, restartabilty and weight. Storeability was considered to be very important since the hopper would not use its fuel until the astronauts came to retrieve them. This ruled out fuels such as liquid Hydrogen and liquid Oxygen. Liquid Hydrazine fuel, on the other hand, has very good long duration storage properties. Solid fuels have excellent storage properties also, but they are not restartable. The hopper will need to perform at least two major burns: the first being the vectored launch burn and the second being the controlled landing burn. Midcourse correction burns may also be required. These burns require very fine control and are best performed by a restartable liquid propulsion system.

Initial sizing of the propulsion system was done for a number of horizontal ranges as well as vertical altitudes. The maximum altitude achieved by the hopper is important since the lunar poles are very mountainous and the hopper will most likely be targeted for a crater which may be one to five km deep. The delta V requirements were not very sensitive to maximum altitude variations but they were sensitive to maximum range variations. Figure 12 shows the propellant requirements for various horizontal and vertical distances.

The studies made of the lunar polar geography showed that a one to two hundred km radius area would cover a large portion of the area of interest. Due to the small mas of the hopper, two 93 pound thrust Rockwell motors were selected. These motors were first used as attitude control thrusters in the Apollo missions, but they will provide sufficient performance for the small hoppers. The fuel tanks were thus sized for a maximum range of 150 Km with a maximum altitude of 15 Km. Figure 13 shows the hopper trajectory and a representative polar relief.



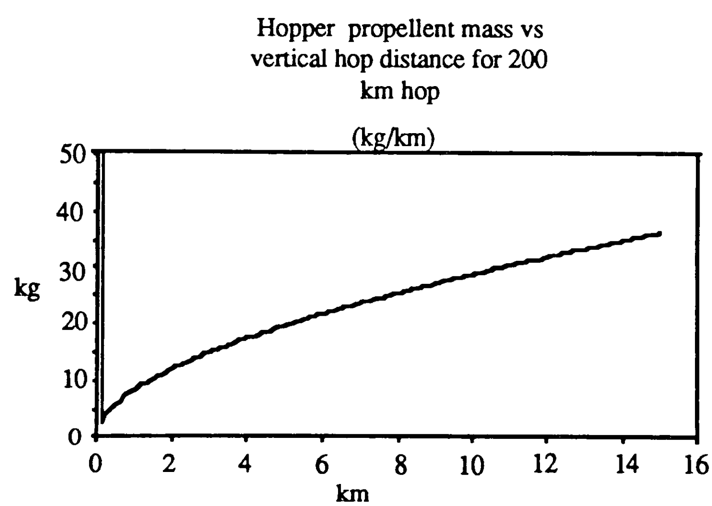




Figure 12: Propellant requirements for various horizontal and vertical distances.

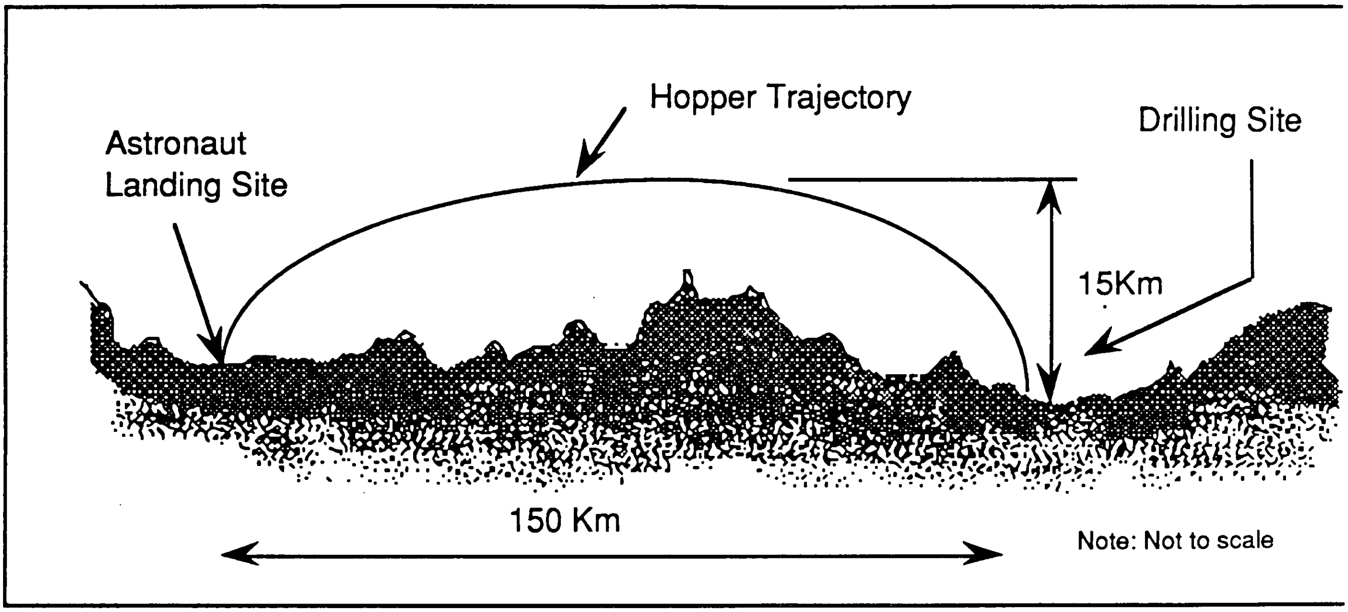



Figure 13: Hopper trajectory and polar relief

## 6.2 Landing Subsystem

### 6.2.1 Landing Dynamics

The lander will be orbiting 110 km above the surface of the moon just prior to the landing sequence initiation. Its orbital velocity will be approximately 1.62 km/s. The first delta V maneuver will be the firing of the main retro rockets to deorbit the lander. This delta V will remove nearly all of the lander's tangential velocity as well as vertical velocity. This burn will begin at an altitude of 110 km and will end when the lander is 10,000 m above the surface. A solid rocket motor will be used for this maneuver. When the burn is completed the main retro rocket propulsion module will be jettisoned from the lander to decrease the touch down mass. At 10,000 m, the vernier retro rocket motors will stabilize the lander and bring it to the surface of the Moon. The vernier retro rockets will decelerate the lander at such a rate that when the lander is approximately 10 to 20 meters from the surface, the lander's velocity will be nearly zero. At this point the rockets will bring the lander to the surface at near hover velocities, 0.5 to 1.0 m/s, and will cut out when all three of the landing pads have touched ground.

### 6.2.2 Probe Landing Gear

The landing gear will be a very important aspect of the probe. The landing gear's most pressing task will be to ensure that the lander touches down in a nearly vertical position. To do this, the landing gear will need to be highly flexible. Among the options studied, conventional legs presented the simplest and and most proven design option.

Figure 14 shows the basic design of the landing gear and its integration with the frame.

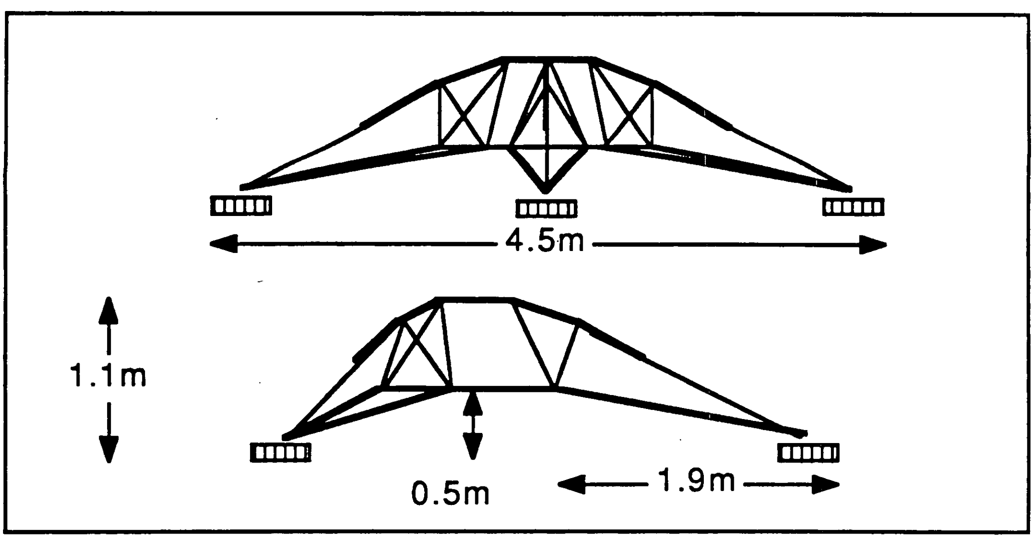



Figure 14: Lander frames and landing gear integration.

The focus of this design is the upper strut of the legs. The upper strut is a passive/active telescoping member which works as follows: under baseline conditions the member is locked (it must be locked during landing since the attitude control motors are mounted at the ends of the legs which must transfer their force) and is not allowed to telescope in or out. When a command is sent by the central processing unit of the lander, this member is unlocked. When unlocked, the upper arm is completely free to telescope in or out. When a second signal is received from the CPU, the upper arm is re-locked firmly into whatever position it finds itself in. Figure 15 shows the landing gear articulation.

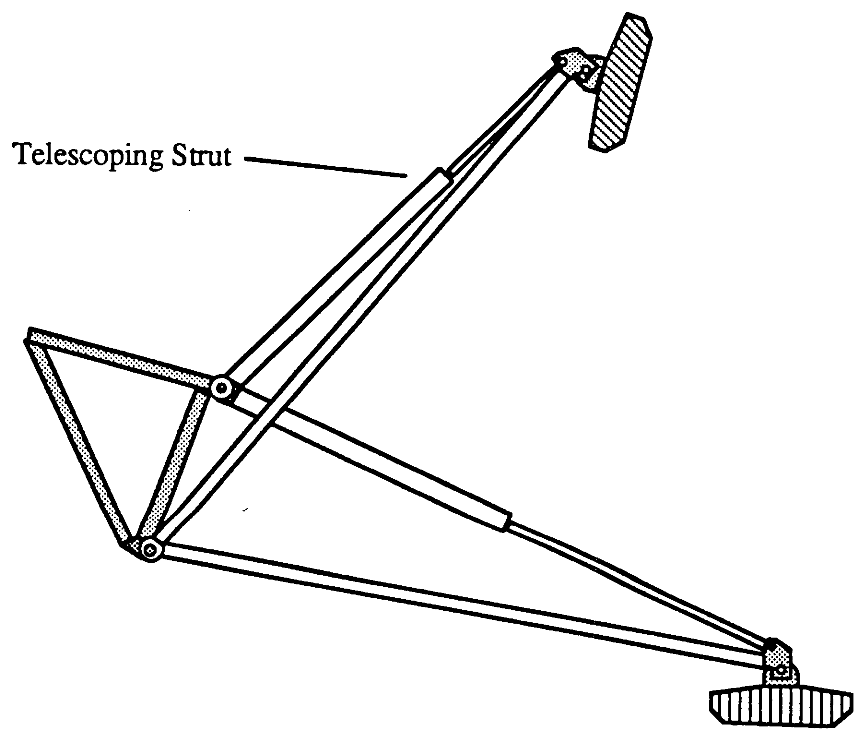



Figure 15: Main landing gear articulation.

This passive/active configuration allows the landing gear to adapt to the unpredictable surface irregularities. The driller can thus be landed in a vertical, or near vertical position without the use of a heavy and expensive active suspension system.

Since the landing gear base will be approximately 4.5 meters wide it will be necessary to fold or somehow store the legs during transport. This is especially important if multiple probes are to be launched in a single vehicle. Two options studied for landing gear stowage are shown in Figure 16. The first option involves hyper-extension of the upper telescoping member to bring the landing gear down. Since the landing gear design is not an active one, (i.e. does not use any actuators) the legs will have to be spring loaded to bring them up to their normal landing position. In such a case, the legs would be held in the folded position by locking the leg's upper telescoping member in the hyper-extended position. Upon deployment of the lander, the legs would be released and allowed to extend to their landing position. The second option involves folding the legs by unhitching the "shoulders" of the landing gear. The legs would then fold downward into their stowed position. Unfortunately, this design requires more moving parts and is therefore considered to be more prone to failure. Option one has therefore been selected as the landing gear stowage option.

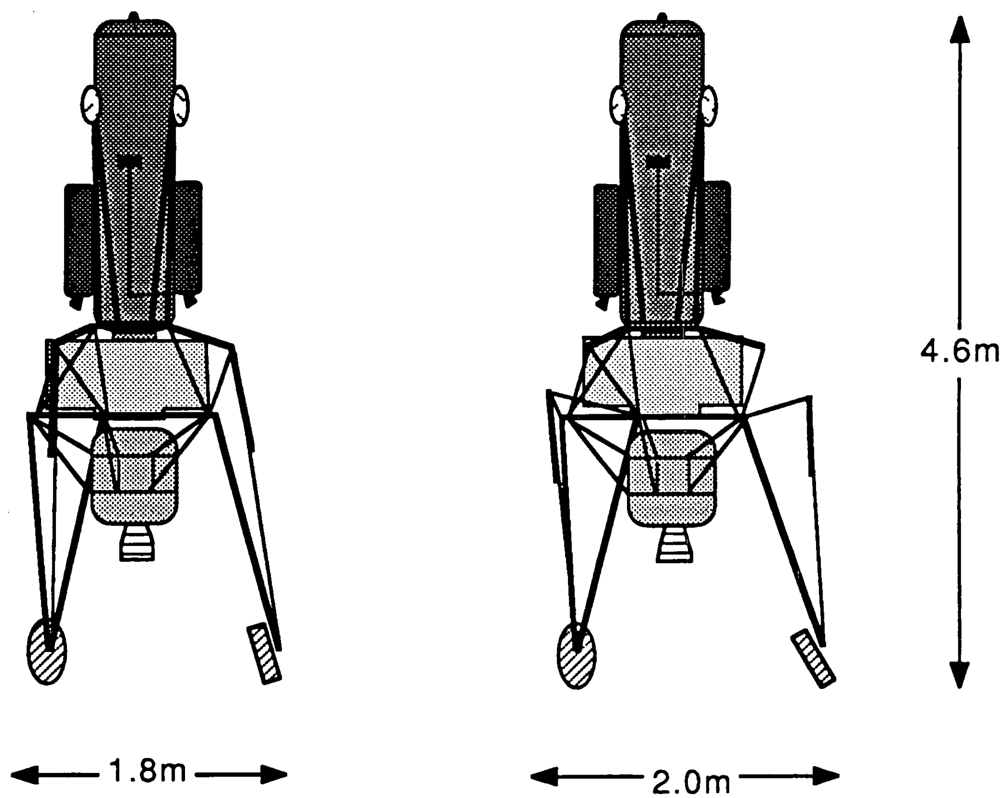



Figure 16: Landing gear stowage options

### 6.2.3 Touch Down

Touch down will be an extremely crucial part of the mission. The lander must land such that its vertical axis is as close to vertical as possible as shown in Figure 17. This is necessary to ensure proper drilling. A semi-passive system has been designed which will ensure proper landing attitude. The system combines the articulated landing gear, a central processing unit coupled with attitude sensing devices, and the propulsion system.

The system works as follows: as the lander reaches the terminal portion of its descent, the computer begins waiting for input from the pressure sensors at the bottom of the landing pads. As soon as a signal is received from a landing pad, the computer commands the upper telescoping arm of the leg associated with that landing pad to unlock. This effectively makes that leg free to move upward without resistance. The lander continues to descend slowly as it maintains its vertical attitude via the attitude control thrusters. When a second leg signals that it too has touched ground the computer unlocks its upper arm also. Finally as soon as the third and final leg signals that it has touched ground the computer commands the upper arms to re-lock in their current position. The lander has now landed safely as well as vertically. If however, the third and final leg did not touched ground before either leg one or two had reached its full range of articulation, the lander would have tipped and fallen over. To preclude this, the command loop of the landing system checks the percentage of the full articulation of each leg continuously. As soon as any one of the legs reaches its nearly full articulation the computer commands the vernier rockets (which are still on, since they are controlling the descent velocity) to thrust the lander upward and outward. This effectively launches the lander away from an inevitable failed landing and allows it to try to re-land in a nearby location. This sequence may continue until no more vernier fuel remains. However, if the landing site is targeted properly, this final ejection hop should be unnecessary. The control loop is illustrated in Figure 18.

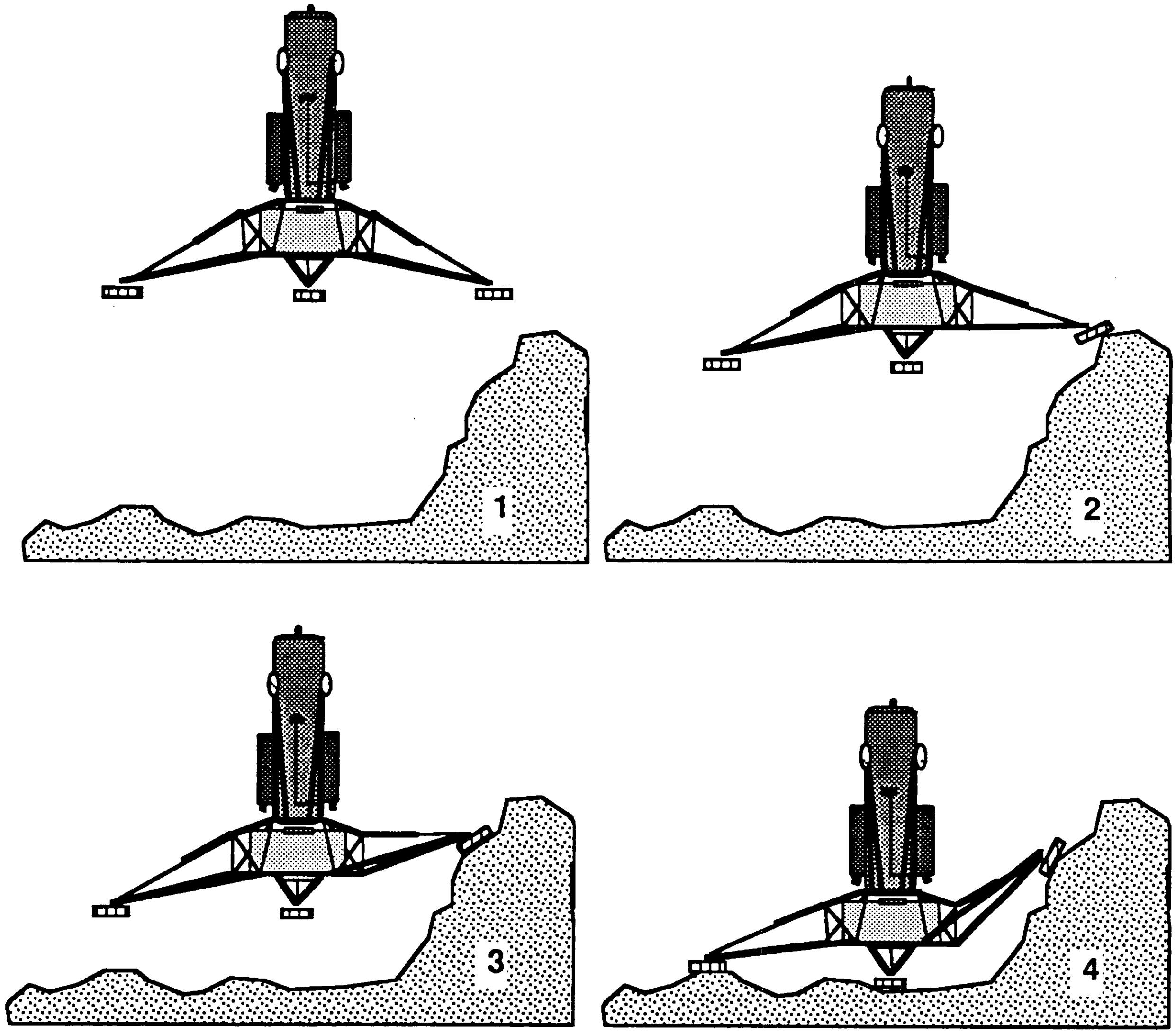



Figure 17: Potential landing scenario illustrating the adaptability of the landing gear to uneven surfaces.

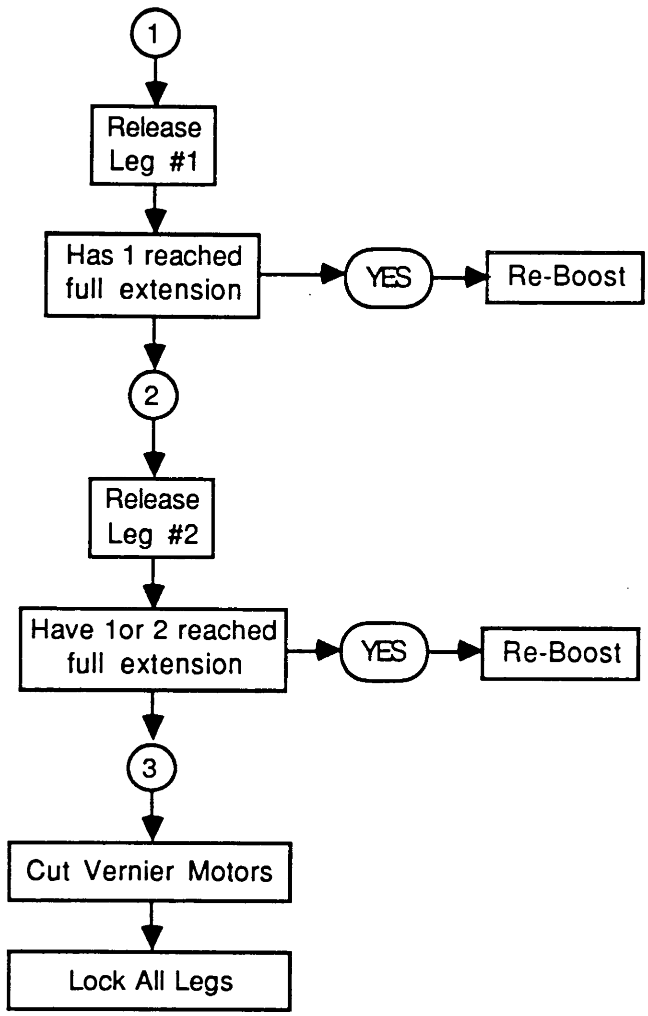



Figure 18: Landing system control logic.

### 6.2.4 Hopper Landing Gear

The landing gear for the hopper will have fewer and less stringent requirements. Its main purpose will be to land the drill core cannister safely at the designated landing site. Since the navigation and control system of the hopper will not be very sophisticated, the landing gear will be designed to have a large base area. The larger base area will increase the hopper's stability on landing, hence allow for a larger margin of error. The force of impact at landing will be absorbed by collapsible suspension members. A preliminary design of the articulation of the landing gear is shown in Figure 19.

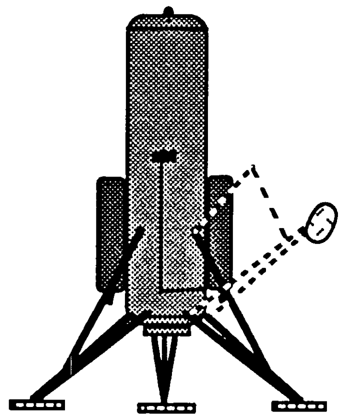


Figure 19: Preliminary Hopper Landing Gear Design

## 6.3 Coring Systems

### 6.3.1 Traditional Drilling Methods Survey

A survey was conducted of traditional drilling systems and techniques which are of specific interest to this project. Specific applications to the Lunar drillers are discussed in the next section.

<u>Drilling operation requirements:</u>

1) Penetrate the subsurface strata.
2) Drill deep enough to reach desired target depth

3) Drill a sufficient size hole for purpose needed
4) Keep the hole oriented in the desired direction
5) Prevent the caving of penetrated strata
6) Be able to retrieve the core samples

The following subsystems are needed in a typical drilling/coring operation.

1) a power plant and transmission system;
2) a hoisting system;
3) a rotating system;
4) a circulating system.

### 6.3.2 Power plant and Transmission System

On Earth, the power plant may consist of several large engines to facilitate not only the drilling system but also all other systems associated with the drilling rig. The power is transmitted to other systems through clutches which drive torque converters. The engines also generate electrical power which in turn drive electrical motors on the other systems such as the rotating, hoisting and circulating systems.

There are other situations in which a full size drilling operation is not needed. Such situations, as the sampling of a shallow well or when a surface core is needed, electric drills can be used to penetrate the surface.

### 6.3.3 Hoisting System

The hoisting system is used in conjunction with the rig to raise and lower the drill pipe in the hole and to maintain the desired weight on the drill head. The entire system is operated by a series of cables and pulleys for the retrieval phase of hoisting.

### 6.3.4 Rotating Equipment

The rotating equipment is used solely to provide a rotating motion to the drill bit and stem. The system includes a rotary swivel, a kelly, and kelly bushing, a drill pipe, drill collars and a drill bit. Power is transmitted by a gear-driven rotary drive. The kelly and the kelly bushings are components which transmit torque from the motor or engine, to the drill pipe. Below the kelly is the drill collar and the drill or coring bit.

### 6.3.5 Drill and Coring Bits

There are several types of drill bits that are currently used in industry. Generally they are categorized as being rotary, percussion or rotary percussion bits. They can be further categorized as being diamond or tungsten carbide bits. Each has unique characteristics in different rock types. Rotary drill bits have rotating heads which bear on the rock and do the actual drilling. The heads are of two basic types, with either milled teeth or with tungsten carbide inserts which break the rock by intrusion, pressure breaking and dragging. Toothed bits are used to drill mostly soft to moderately hard formations. Whole insert bits are used on moderately hard to very hard abrasive surfaces. For extremely hard formations and for some coring operations, coneless bits with industrial grade diamond inserts are used. Although they are more expensive, diamond bits are often capable of a considerably higher penetration than insert-type core bits.

Figure 20 shows the basic design of three types of diamond bits which are commonly used. Figure 21 shows two types of tungsten carbide insert bits which are also commonly used.

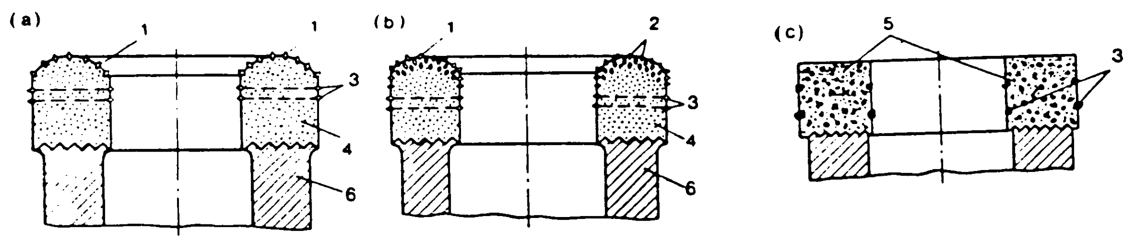



Figure 20: Types of diamond bits. (a) surface-set (single layer) bit; (b) three layer bit; (c) impregnated bit: 1- face diamonds; 2- face diamonds of second and third layers; 3- reaming (gauge) diamonds; 4- annular matrix;5- matrix impregnated with small diamonds; 6- bit body (Reedman , 1979)

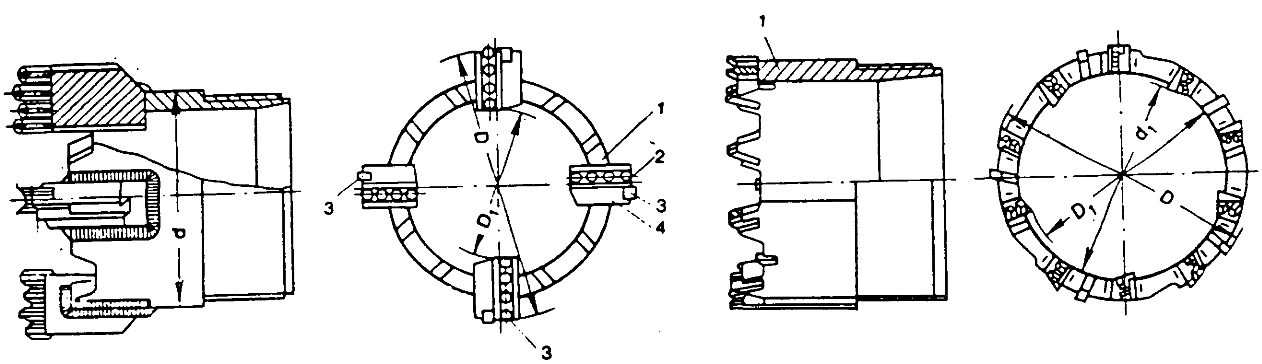



Figure 21: Tungsten Carbide-insert bits. 1- bit blank; 2- main (face) cutter; 3- reaming (gauge) cutter; 4- setting plate

### 6.3.6 Circulating System

The circulating system is used in terrestrial applications to lubricate the drill string and to remove the cuttings from the core hole. Lubricants are forced into the hole to displace the cuttings and force them to the surface. The circulating fluid (almost always water) is then screened to sift out the large debris before returning to the hole to complete another cycle.

### 6.3.7 Traditional Coring Methods

A survey of traditional coring methods was conducted. The results are summarized here in general. Specific applications to a lunar drilling probe is discussed in the next section.

### 6.3.8 Coring and Core Analysis

The definition of a core is any sample of subsurface rock strata obtained from a borehole and of a size large enough to permit the measurement of its important physical properties. In this respect, the core offers the only direct means by which a desired rock can be evaluated in terms of its physical characteristics, specifically its fluid content and its fluid-flow properties.

Many physical, chemical, and thermodynamic properties can be determined by core analysis, but the most important pieces of information normally obtained from a core analysis are porosity, permeability and fluid saturations including water, oil, and gas. Cores are also obtained for a variety of other reasons, including lithological and other types of geological studies, studies of rock fracture patterns, and studies to define or improve drilling practices.

### 6.3.9 Coring Methods

Many types of equipment and many different techniques have been devised for obtaining core samples. Current methods of coring can generally be grouped into conventional, wireline, diamond and sidewall coring.

#### *Conventional Coring*

Conventional coring techniques are defined by the type of cutting action that occurs. There are three types of cutting action in conventional coring: rotary coring, percussion coring, rotary-percussion coring. The rotary coring is used in soft materials. The bit is forced downward with an applied axial force and rotated causing a helical pattern impressed upon the surface. The percussion technique causes a high impact exchange of energy from the bit to the rock which causes the breaking of the rock mostly identified as the "jackhammer" technique. The final option is the combination of the previous, the rotary-percussion. This method causes the breaking of the rock with a percussion motion and clears the debris by using the rotary motion.

The coring assembly for conventional rotary coring consists of a coring bit and a core barrel which are both located at the end of the drill stem. The coring bit is designed essentially the same as are standard rotary rock bits, except that it has cutting surfaces only on the perimeter. The core barrel is designed to receive and retain the core in the core barrel as the bit drills into the rock. Typically, the core barrel consists of an inner receiver barrel, an outer barrel, a core retainer or catcher and a pressure relief valve to vent core barrel pressure to the outside of the drill stem. Figure 22 shows a wagon drill arrangement which is commonly used in follow-up work on geochemical soil anomalies.

Drilling fluid circulates between the inner and outer barrels in order not to flush the core, thereby increasing recovery of the core. For the same reason, some core barrels have a free floating inner barrel, which is free to rotate or remain still. For use in very soft, friable or unconsolidated rock, the inner barrel may consist of a heavy rubber tube, or sleeve, which is removed at the surface with the cored formation intact on the inside of the tube.

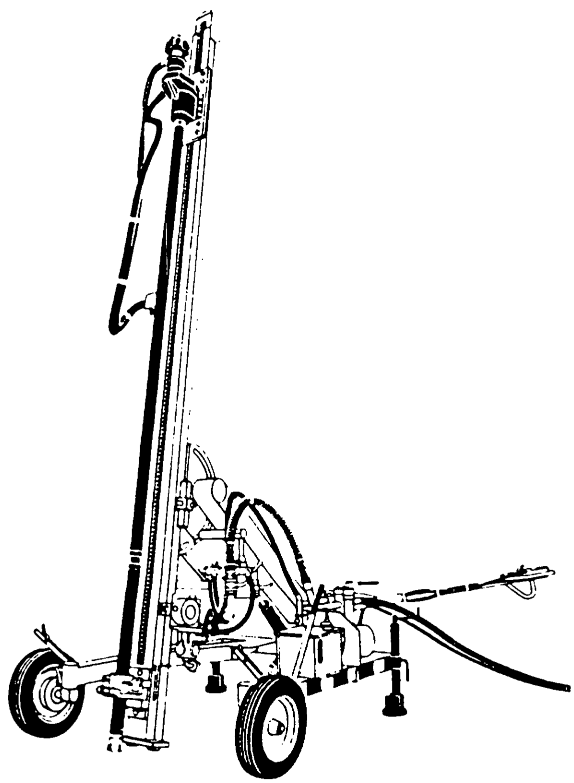


Figure 22: A typical wagon drill used for shallow core retrieval.

The chief advantages of conventional coring is that a large diameter core of up to 75 cm can be obtained and with the proper techniques, generally good recoveries can be expected. The chief disadvantage of this method is that the drill string must be removed to attach the core barrel and bit and then removed again to recover the core and resume normal drilling

### *Wireline Coring*

This type of equipment permits intermittent coring and wireline retrieval of cores, all without interrupting the drilling operation. All that is required is the installation of a core bit on the end of the drill stem. When a core is to be taken, a retrievable core barrel can be pumped down the drill pipe to a locking seat at the drill bit. After the core has been cut, the core barrel assembly with the core can be recovered by using a wire to hoist the core up through the drill string.

When normal drilling is required, a drilling center bit assembly can be pumped down to seat and lock in the core bit, thereby providing a full cutting head. This can be removed with a wire-line when another core is desired.

The major disadvantages of this method are that only a small diameter core can be obtained, and core recovery is not as good as conventional coring. The major advantage is that a core can be taken from a hole without having to remove the entire drill string. Figure 23 shows a typical wire-line roller core bit.

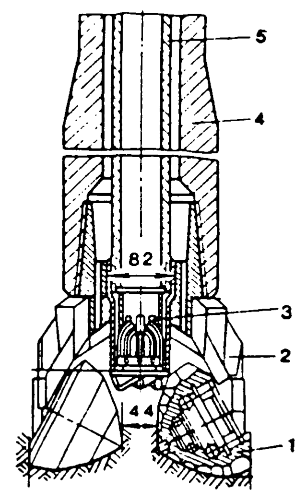



Figure 23: Wire-line roller core bit.
1- roller cutter; 2- bit head; 3- core lifter; 4- bit body; 5- wire-line core barrel

*Diamond Coring*

This method is used to increase core recovery and coring rate, primarily in hard, dense formations. It is generally the same as conventional coring, except that the coring bit is faced with a hard metal matrix in which a number of industrial-grade diamonds are imbedded.

The major advantages in the use of diamond bits are the faster rate of penetration and a longer bit life, permitting a core of 30 meters to be taken before removing the drill stem. Figure 24 shows a typical double-tube core barrel configuration for diamond coring.

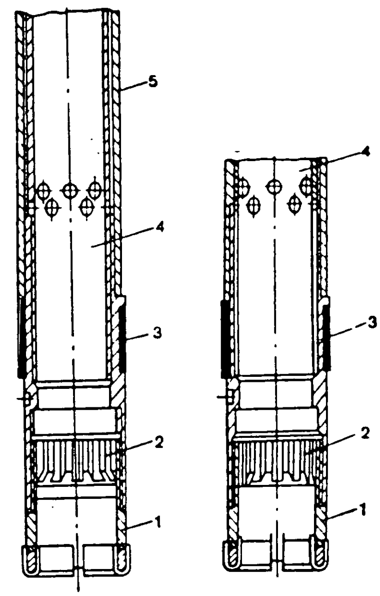



Figure 24: Double-tube core barrels for diamond drilling. (a) ridgid barrel (b) swivel barrel; 1- diamon bit; 2- core lifter; 3- diamond reamer shell; 4- inner barrel (tube); 5- outer barrel

*Sidewall Coring*

This is a supplementary method used to obtain core samples from existing holes. This tool is designed somewhat like a perforation tool in that it drives a short tube into the side of the borehole. Retrieval is then made by means of a cable attached to the body of the tool (Austin,1983).

The advantage of this method is that samples can be taken at any depth after a hole is drilled. The disadvantage of this method is that the samples are small, approximately 25 mm in diameter and 50 mm in depth. Also, the core is usually disturbed and contaminated by the drilling device, which often uses gunpowder to drive the sampling tube into the wall of the hole.

### *Other Methods*

- Flame and Electric Arc
- Chemical Softeners
- Abrasive Jets
- Explosive Shape Charges
- Lasers
- High Frequency Vibration

Many other drilling methods exist but a large number of them are not applicable to lunar drilling. Some which could be used on the Moon have the disadvantage of contamination hence are not acceptable for this mission.

## 6.3.10 Lunar Adaptation of Drilling and Coring Methods

The lunar drilling systems have different requirements than typical drilling systems on Earth. Therefore several modifications have to be made in order to modify traditional drilling methods. Considerations are made due to the following:

1) Lunar gravity is 1/6 of the earth's gravity
2) Lack of atmosphere (colder temperature, vacuum)
3) Surface conditions (120 K, unknown rock density, unknown materials to be drilled)
4) Constraints (No liquid lubricants or circulation)
5) Autonomous drilling operation

### 6.3.11 Drilling operation requirements for the lunar driller

1) Penetrate the subsurface strata.
2) Prevent the caving of penetrated strata
3) Prevent any debris or gases from escaping
4) Drill deep enough to reach desired target depth
4) Drill a sufficient size hole for purpose needed
5) Keep the hole oriented in the desired direction
6) Be able to retrieve the core

### 6.3.12 Coring Method for the Lunar Driller

***Core Description***

The core samples will be 2.5 cm in diameter. This size sample will be large enough to permit the measurement of its important physical properties, yet keep the total mass to be returned to earth minimum. Initial studies showed that the first 3-5 meters of regolith will contain the greatest amount of water, if any at all (Hapke,1975). It was decided to core down to 15 meters since that is approximately the depth which may be of significant commercial value if a permanent or semi permanent base were established at the pole. Although water is the primary substance being sought at the pole. Other minerals or elements may also be found which could be of even greater value to humans.

***Coring Methods***

Considering the methods of coring presented in the survey, all but the rotary-percussion technique have been eliminated because the necessary constraints (ie. contamination of the core, power consumption) have not been met.

Among the traditional coring methods which were surveyed, the most suited technique is the rotary-percussion due to its combination of axial crushing and rotary removal of the debris. Another advantage of the rotary-percussion method is the fact that it produces a lower bit temperature when coring compared to the diamond rotary method. The coring assembly for conventional rotary percussion coring consists of a coring bit and a core barrel both located on the end of the drill stem. The core barrels for the Lunar lander will be a full length hollow tube which is divided into 10, 1.5 meter sections designed to receive and retain the core as the bit drills into the rock.

### *Hoisting System*

Due to the bulk and mass constraints, the driller will not contain a large drilling structure for hoisting purposes. The hoisting will be provided by the same mechanism used to drive the drill string into the moon. The system will simply be put in reverse. During retrieval, the rotary motion of the drill stem will be maintained to reduce friction. The percussion action will however be ceased during this process.

### *Forcing and Rotating Equipment*

The rotary percussion method requires both a rotation of the drill string along with a reciprocating downward axial force on the drill bit. The axial force (220 Newtons Lunar gravity) will be provided solely by the driller's weight which is more than sufficient. The percussion movement is generated by a rotating, spring loaded cam.

The drill string is loaded into a kelly device which is driven by the motor. The kelly rotates and allows the drill stem to pass through it while it applies a constant torque. This rotation causes the cutting, or coring action to take place at the end of the string while forcing the drill chips out.

### *Bit Selection*

The purpose of the core bit is to cut through the rock strata and reach the target depth in the shortest amount of time. Since time is crucial to the lunar driller's mission, the emphasis on rate of drilling is a high priority in the drill bit selection.

There are disadvantages to using the diamond bits which make it necessary to disqualify these bits from the mission bit selection. Research showed that the diamond bits are limitation by heat constraints, inability to sustain a percussion force and tendency to dull in very hard rock formations (Crouch, 1965). The diamond heads become brittle with an increase in temperature. Without a cooling process, the diamond bits break off. To consider the diamond bits, it would be necessary to incorporate a temperature sensor in the drill bit to sense the high temperatures. As the bit temperature approaches that of the water's sublimation point, the driller would need to cease operations to let the drill head cool. Since the temperature of the surroundings are at a constant 120 K, the cooling time will be minimal. The diamond bit will also begin to dull after drilling without lubrication. Because of these disadvantages, the diamond head bit was not selected.

Another core bit option is the Tungsten Carbide bit. This bit is more desirable in operations which require percussion as well as rotation. The advantage of using a Tungsten Carbide bit is the fact that it is able to withstand the high temperatures caused by the friction of dry rotary-percussion coring without dulling.

Preliminary studies show that the conventional coring method of rotary-percussion technique with a Tungsten Carbide bit appears to be the optimum combination of choices. This method will increase core recovery and coring rate, primarily in hard, dense formations.

### 6.3.13 Driller Design

The lack of manned intervention requires the driller to be fully automated and very robust. Furthermore, it must be able to drill and core in variable rock types as well as withstand the harsh environment.

***Driller Breakdown***

- The motor(s)
- The drill string
- The "skirt"

The motor(s) on the Lander will have several functions. These functions include rotating and driving the drill stems, transferring the torque, and supplying the percussive motion. In order to minimize the weight and power consumption of this critical part of the probe, a single motor will be used.

Mechanical gearing will allow the driller to share its power between torque, axial force and percussion. Gears will also be used to vary the drilling rotation rate. The lubrication of these parts is a point of concern. Any liquid lubrication would be difficult and hazardous as it may contaminate the samples. This would also require pressurization, circulation, isolation, etc. Thus, it is recommend that the gears and clutches be fabricated with self lubricating materials or solid lubrication coating (sulfur steel, silicon coating).

Research for the Apollo coring experiments (Crouch, 1965) showed that D.C. motors had excellent power to weight ratio and also proved to be very reliable. The operation of the drilling system requires that the drill string serve not only as a core containment device but also as a force transmitter. The motor creates a torque which is transmitted to the drill string which in turn transmits a rotation motion to the bit. A second force which is transmitted through the drill string is the percussion motion.

The drill string must also contain the core during the coring process and is the means by which retrieval is possible. The drill string sustains a variety of stresses under such conditions as: cryogenic temperature, high abrasion from the drill walls, dust, lack of lubrication, impacts from percussioning. However, the string is transported from Earth and must be light weight. Therefore, it is necessary to use a material that will withstand this harsh treatment. Titanium is among the best materials for this purpose. The string is composed of ten 1.5 meter Titanium stems. The geometry of the drill stem is seen in Figure 25. The internal diameter of the drill stem is 2.5 centimeters and the external diameter will be no more than 3 centimeters.

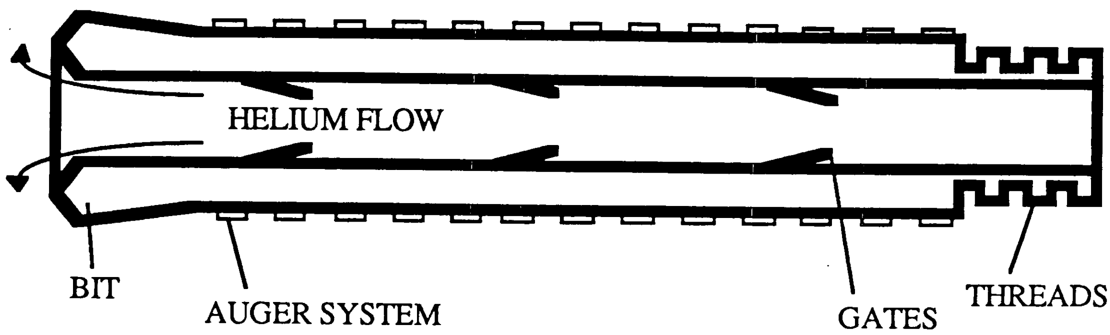



Figure 25: Drill stem geometry

As the drill string bores through the regolith, contact with the walls may cause an undesired build up of friction. Therefore, it is necessary to reduce the friction between the string and the drill walls. To accomplish this task, the bit will be slightly larger in diameter than the stems. This creates a space between the drill stem and the drill wall which decrease the friction caused by contact. This potential problem must also be dealt with by using some form of anchoring. This should prevent the lander from slipping during coring operations.

Since the probe will be entirely automated, the operations imposed upon the driller must be as simple as possible. The stems will be stored in the upper part of the probe, the hopper. These stems will be contained in the circular arrangement seen in Figure 26. When a stem has been completely deployed, the next stem is positioned for insertion by rotating the turret and releasing the stem into the drill press where it is attached to the preceding stem. During this transition time, the vertical and percussion motions are suspended. Following this process, the drilling operation resumes. A detailed description of this turret feeding system can be found in the Hopper section.

During coring operation on Earth, many trapped subsurface gasses escape through the drill hole. In the event that such a phenomena were to occur on the Moon, it would be very important to sample these gases and to analyze their

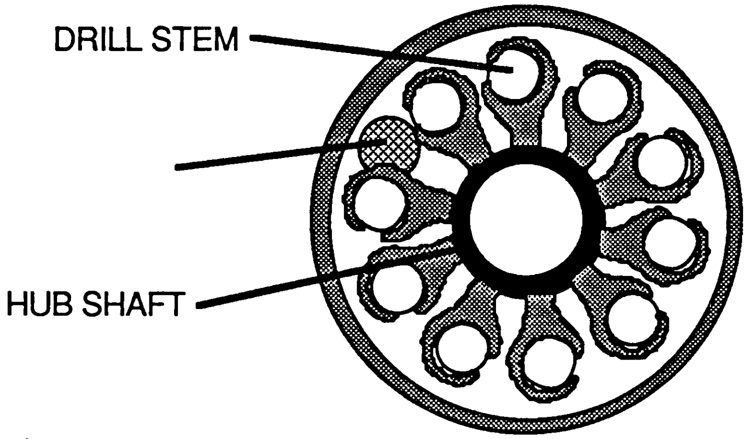


Figure 26: Drill stem storage turret

composition. To do this, a skirt has been designed which will surround the drill string as seen in Figure 27. The skirt will serve only to momentarily contain these gases so that a small spectrometer under the lander can analyze the gas.

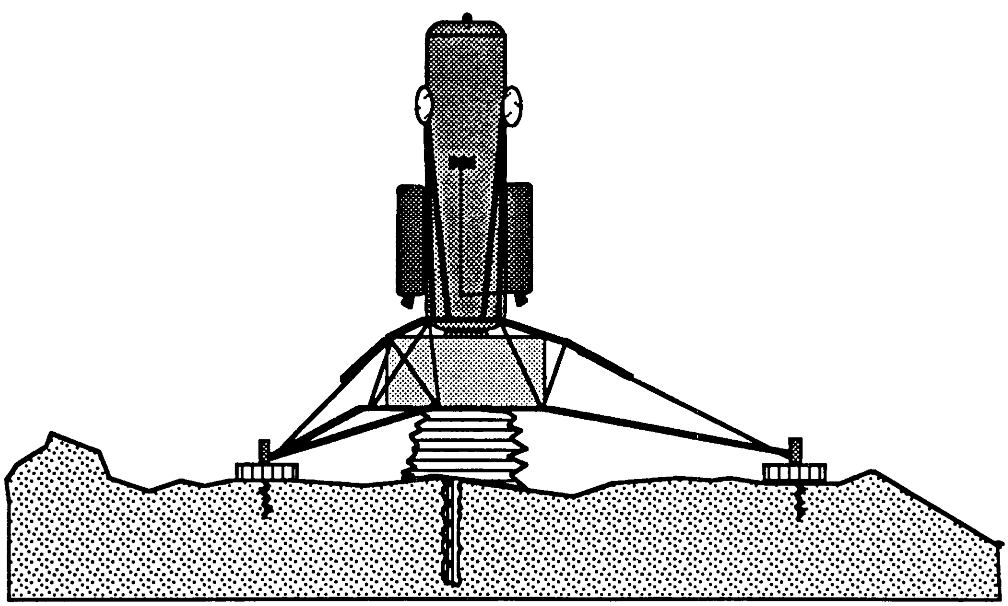


Figure 27: Lander with skirt used for volatile emission containment.

*Drilling process*

The drilling process is divided into the following sections.

Drilling Automation
Chip removal
Core retrieval

There are several parameters that can be varied in order to optimize the drilling process specifically to increase penetration rate and decrease power consumption. The rotation speed can be adjusted to gain the maximum penetration rate depending on the hardness of the regolith. Also, the amplitude of the percussion can be varied to account for changing rock density. In soft rocks, slow pure rotary is recommended but faster percussion is mandatory for hard rock.

The temperature of the drill bit increases as a function of time, rate of rotation and axial loading. Once the bit has reached a critical temperature, coring must be stopped to allow for the bit to cool. This will be a minimal time due to the low lunar temperatures. Figure 28 is a block diagram which illustrates the control process necessary to automate the coring.

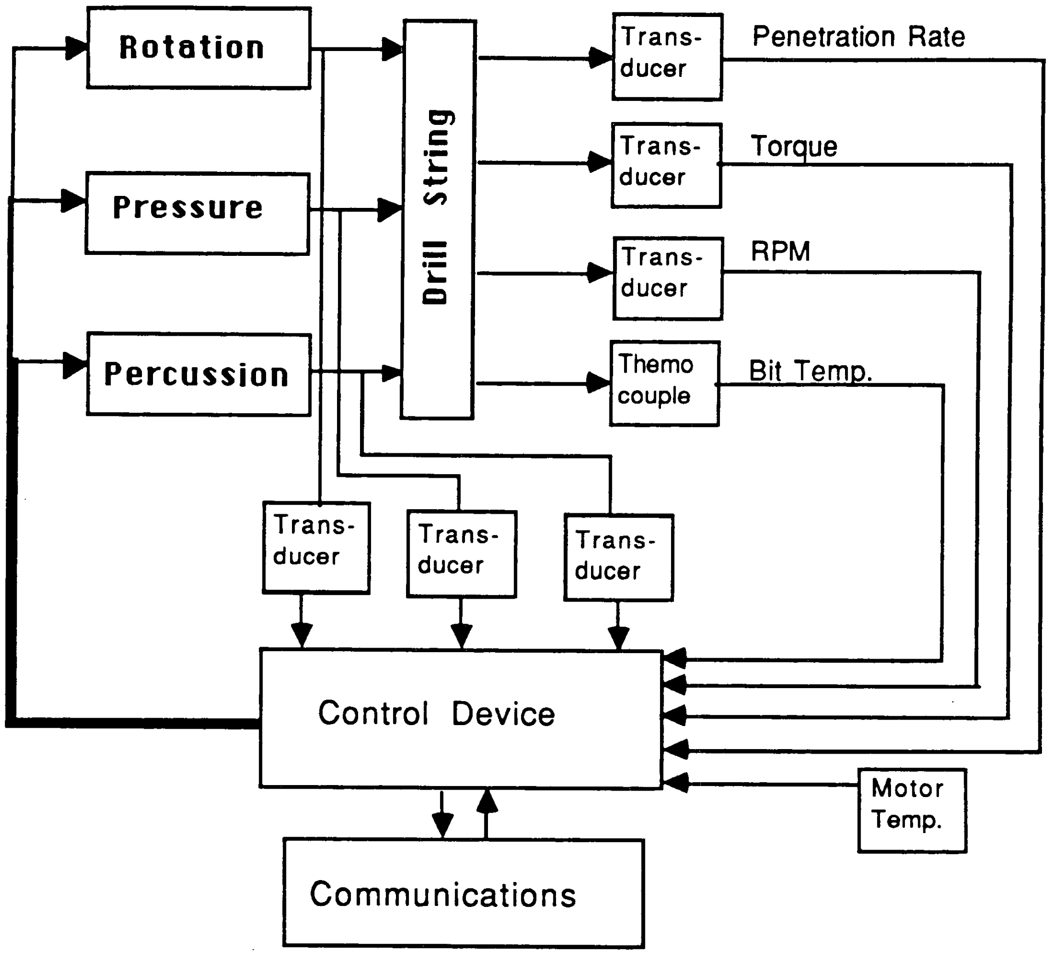



Figure 28: Automation block diagram

On Earth, the problem of removing drill chips or displaced soil is solved by using a continuous flow of water or mud through the drill hole. Obviously, fluids cannot be used on the lunar surface because of the lack of an atmosphere. One debris removal method incorporated into the design of the drill section is the use of helical augers on the outer wall of the stem. This will push the dust and debris from the bottom of the hole outward. Research, conducted for the Apollo Lunar Coring Apparatus, suggests that the auger system may jam at depths greater than 10 meters (Crouch,1965). To overcome this problem, helium gas could be blown down the hole, forcing the drill chips out . This flow of helium will also cool and lubricate the drill bit, improving the quality of the drilling. The Helium flushing will be done intermittently. As each drill stem reaches its full penetration, and before the next stem is attached, a burst of Helium gas will be blown down the string. The combination of the auger and the flushing gas has been incorporated in the final design. It is estimated that approximately 20 liters of debris will be created in coring a 15 meter hole.

The retrieval of the core is among the most important parts of the mission. Therefore, the reliability of this operation must be as high as possible. The first option for retrieval makes use of an inner cylinder inside the drill stem. This light and self lubricating cylinder (carbon composites, Teflon, etc.) would be pulled out of the drill string, leaving the string in the hole. This option would also reduce the friction of the core coming into contact with the drill string. A disadvantage is the fact that the process of connecting the inner and outer drill stem sections is very complicated. The second option considered would retrieve the entire string. This method is a much simpler design but requires more mass to be lifted. This option also requires several one-way gates at the bottom of each section to prevent the core from sliding out the bottom during the retrieval. With this method, it would be convenient to leave the bit at the bottom of the hole thus reducing friction on the outer wall during the removal process. Figure 29 shows the lander while the coring is in progress.

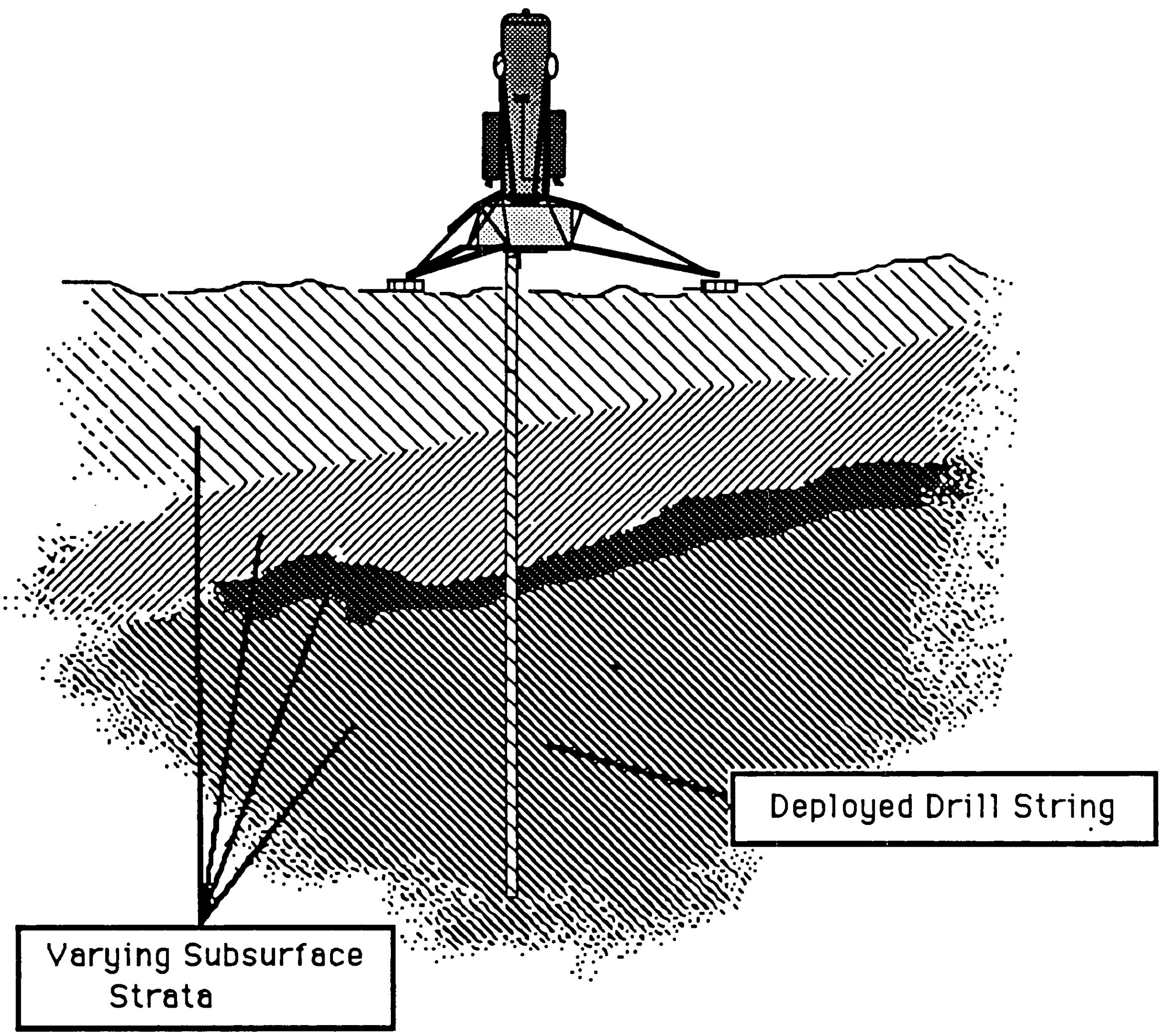


Figure 29: Lander coring operation in progress

### 6.3.14 Logging techniques

Any type of core analysis of a region is limited to the actual size of the core sample taken. Since there are obvious limitations to the maximum size of a core sample, other methods must be used to "see" further into the soil. Such a technique is known as logging. Logging involves lowering a sensor into an existing hole and taking indirect measurements of the surrounding soil. There are three main types of logging:

- Electromagnetic wave detection
- Magnetic resonance
- Neutron gamma emission

For the specific application of water detection, the molecular resonance of the water molecule is used. Water molecules have a natural resonance frequency of 1.1 GHz. If exited at this frequency water molecules can be detected with very sensitive equipment. Magnetic resonance is especially accurate in detecting Hydrogen atoms from their response to an artificially induced magnetic field. Finally, neutron emission is used to interact with the surrounding soil. Neutron emission stimulates gamma ray emission from specific molecules. These gamma rays are then detected and soil content can be inferred. All of these techniques are feasible and allow a much larger volume of regolith to be sampled. Whether or not this technology can be applied to a fully autonomous lunar probe is a matter which will require research.

### 6.3.15 Anchoring Device

Two factors come into play during coring operations which could require the use of an anchoring device. These are axial bit pressure and maintenance of proper drill-hole alignment. The coring operations has been designed to operate at a nominal axial bit pressure of approximately 220 Newtons. This corresponds to a 1320 Newton force on Earth. Since the weight of the lander on Earth will be over 2500 Newtons, it will not be necessary to use an anchor to provide any axial force.

If the drill stem and the whole become misaligned during coring, the power consumption will increase markedly and the penetration rate will drop off very quickly. It will therefore be essential to provide a method to prevent the lander from moving while coring. Two main problems arise immediately in considering an anchoring technique. The first being that the soil beneath the lander may either be solid rock, loose soil or both. Each one of these conditions requires a different type of anchoring device. The second being that the system must be completely self deploying.

Since it will be impossible to know ahead of time what type of ground conditions will be found at the landing site, it will either be necessary to design two separate anchoring systems or somehow design one to work in any soil condition. Oasis has designed the anchoring system as follows:

A total of three anchors will be used, one at the end of each leg. Each anchor will have the possibility of acting as a loose soil, soft silt, or hard rock anchor. This is done by placing a sharp thin drill bit at the end of a longer, extendable auger. Figure 30 shows the anchor design.

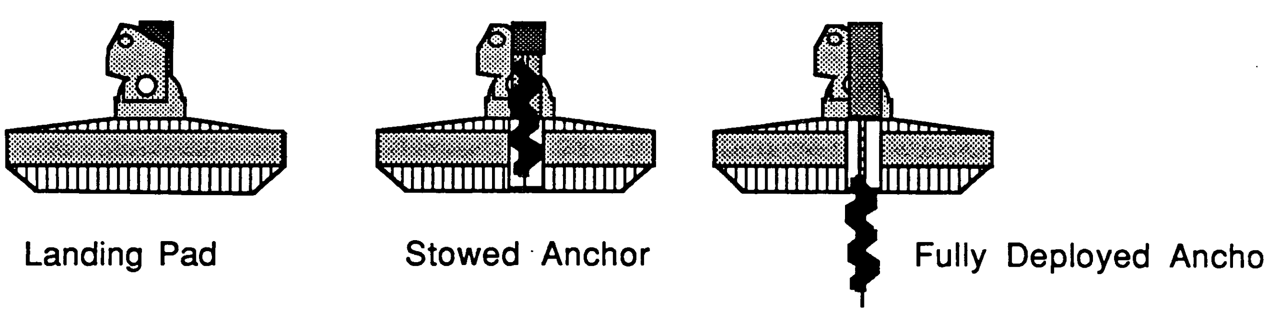



Figure 30: Landing pad with anchoring device.

After the lander has successfully landed at the pole, the anchoring procedure is initiated. First, each anchor assumes the terrain under it is hard rock. It therefore begins drilling a small narrow hole with the end bit of the anchor. If the ground actually is hard rock, the drill bit will bore into the rock and then stop. This will prevent the lander from twisting during drilling. If, however, the ground is not hard rock, the drill will meet no resistance. When this happens the auger will begin to extend outward, hence digging its way deeper into the looser soil.

This system will be sufficient to prevent the lander from slipping, or twisting on itself during coring operations. It is not, however, designed to provide any axial force for drilling. The weight of the lander itself has been determined to be more than sufficient for this.

## 6.4 Thermal Control Systems

The thermal control system necessary for a lunar polar driller is extremely important, since the system will require both heat rejection and heat generation devices. To compensate for the extreme cold temperatures at the poles, estimated to be below 120 K (-153 °C), the driller's thermal control system will be dominated by the need to keep its equipment warm. The following sections describe both the possible heat rejection and heat generation devices.

### 6.4.1 Heat Rejection Devices

The dissipation of heat from the driller as it descends to the lunar surface and during the drilling process, will require heat rejection devices. Since the deployment and descent stage is of short duration, a passive device with low power and mass characteristics would be more appropriate over active thermal control systems. The latter offers much larger heat rejection capabilities, however, with marked power and mass increases. However, to tackle the problem of extreme heating during the drilling process an active device is required to accommodate the large heat load.

Passive thermal control methods, which have been chiefly used in earth-orbiting satellites, typically have low heat load rejection capabilities. However, they may have applications in probes and systems requiring minimal thermal control. A promising passive technique was developed to radiatively and conductively isolate the radiating plate from the spacecraft by means of low-emittance, highly specular angle radiation shields. The object is to increase the radiation cooler capacity by reducing parasitic heat flows through radiative and conductive heat paths into the radiating plate. The shields are placed adjacent to each other with an included angle of 1.5° between adjoining shields, with the wider opening of the shield facing space. Thermal energy is radiated from the shields into the cavity by reflections off the angled shields. Figure 31 shows the angled shield concept (Bard,1981).

The performance of this design was estimated by the use of analytical thermal models, and heat flows were determined for the cooler in a 900 km circular Earth-orbit. A preliminary comparison between this Advanced Radiative cooler (AR), and the advanced temperature sounder (AMTS), and the VISSR flight cooler (launched in 1974) is shown in Table 1 (Bard,1981).

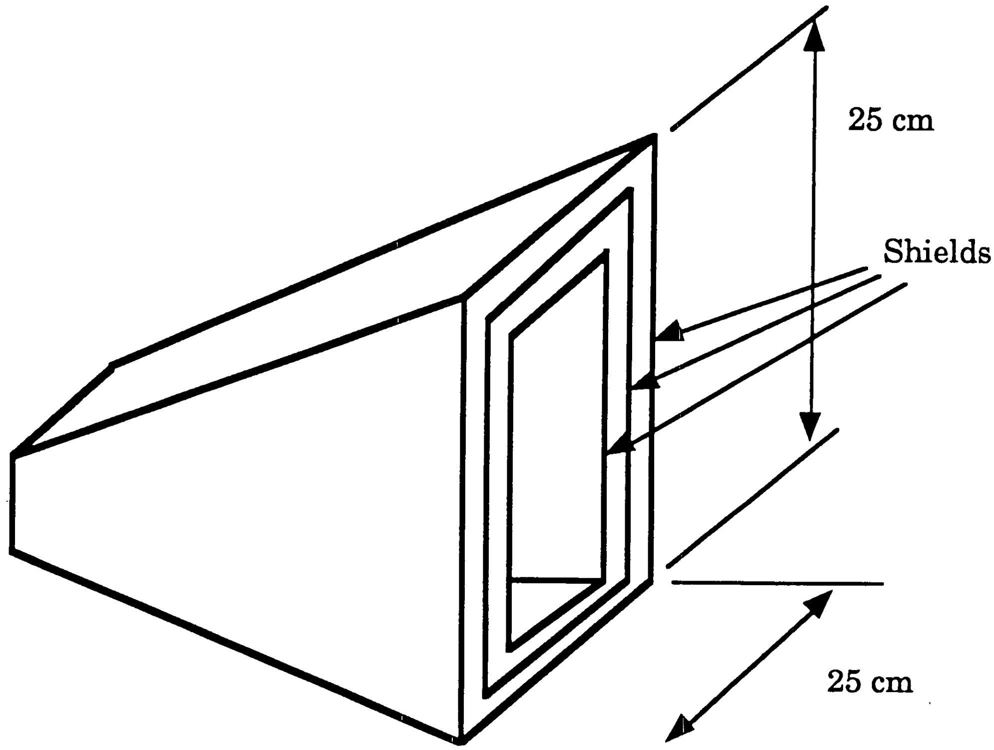



Figure 31: Angled Shield Passive Radiator

Table 1: Passive Heat Rejection Device Comparisons

| | AR | AMTS | AR | VISSR |
|---|---|---|---|---|
| ORBIT (km) | 833 | 833 | 900 | 36,000 |
| MOUNTING TEMP * (K) | 300 | 300 | 300 | 300 |
| RADIATOR TEMP (K) | 64.4 | 64.4 | 81 | 81 |
| HEAT LOAD (mW) | 250 | 250 | 93 | 2 |
| RADIATOR AREA (m2) | .62 | 5.0 | .16 | .16 |
| WEIGHT (kg) | 13 | 45 | 2.2 | 3.6 |

* Planet shade assumed to be at Ts=160 K

When compared to the AMTS, the AR achieves the same heat load and temperature conditions with only 25% of the area and 29% of the AMTS weight. Additionally, in comparison with the VISSR, the AR is only 61% of the VISSR's weight and possesses 5250% greater cooling capacity. Based on this analysis, the advanced angled shield passive radiator would be useful for this mission. The radiator weight and size,to handle a 93 mW heat load, will be 2.2 kg and .16 $m^2$ respectively.

Another concept for passive heat control, which has been used extensively, is the multi-layered thermal blanket. Due to their thermal properties, a great deal of heat is restricted from passing into or out of the blankets. Thus, they can be used to safeguard the driller's subsystems from experiencing extreme temperature conditions. The effectiveness and low mass of the blankets warrant their use on the drillers. The estimated mass for a number of these blankets distributed among the various subsystems is 3.6 kg (Beckman,1986).

Lastly, but very important, on the list of heat rejection devices is equipment necessary to cool the drilling apparatus. The best method to this date, which was investigated by NASA for use by the Apollo drilling mission but never used, is to use a Helium gas cooler system. The Helium will be used to cool the drilling motor and other parts, in addition to serving as a means of flushing out drilling debris. The implementation of such a system as well as the mass and power requirements are unknown at this time since the field of dry drilling is very new.

### 6.4.2 Heat Generation Devices

There are several heat generation devices which can be used to satisfy the need for heat on this mission. A device with possible use is an electric resistance heater, which generates heat by running electrical current through a resistor (Doe,1989). Another possible method for heating is to use a radioisotope heater unit. Calculations show that to raise the temperature from 120 K to the temperature region of 283 K required for driller equipment operation, approximately 352 Watts of power are required -- assuming a total radiator surface area of one square meter (Doe,1989). Using these power and dimensions as a good approximation, it is evident that the requirements for an electric resistance heater are very constricting, therefore a more efficient device is necessary.

The need to keep the driller's components functioning reliably in the cold climate of the lunar pole can also be satisfied with a number of smaller radioisotope heater units. One such unit, which can maintain components at desired temperature levels, is the Light Weight Radioisotope Heater Unit (LWRHU). The LWRHU is a $^{238}PuO_2$ fueled container which is designed to emit one W of thermal power in dispersed locations aboard a spacecraft. This device was also designed to be as light as possible and still provide protection of the plutonium fuel over the unit's lifetime, in all credible accident situations. The heater is a cylinder 32 mm high and 26 mm in diameter, weighing slightly less than 40 g. A summary of the LWRHU specifications are displayed in Table 2.

Table 2: Light Weight Heater Unit Specifications

| Parameter | Value |
|---|---|
| Thermal Power | 1.1 +/- 0.03 W |
| Accuracy of Measure Power | +/- 1% |
| Mass | 40 g |
| Useful Lifetime | 7 years (minimum) |
| Configuration | Right Circular Cylinder |
| Surface Temperature in Free Air | 45° C |
| Impact Resistance on Reentry (intact capsule) | 49 m/s |

Previous space missions -- including Pioneer 10 and 11, Voyager 1 and 2, and earlier mission -- have used similar radioisotope heater units which were fueled with 80% enriched $^{238}PuO_2$. This particular radioisotope is the consensus choice for use in the heater units, due to its long half-life (87.7 years), adequate specific thermal power (0.42 W/g), and tolerable level of penetrating radiation (Tate,1982). Radioisotope power sources have also been used successfully as components of radioisotope thermoelectric generators (RTG's) in the Transit, LES, Pioneer, Voyager and Apollo programs. These isotope systems have developed and outstanding performance and safety record. One hundred LWRHU's were installed on the Galileo spacecraft after extensive performance and safety verification testing (Cull,1987). A detailed drawing of the LWRHU can be seen in Figure 32 (Tate,1982)

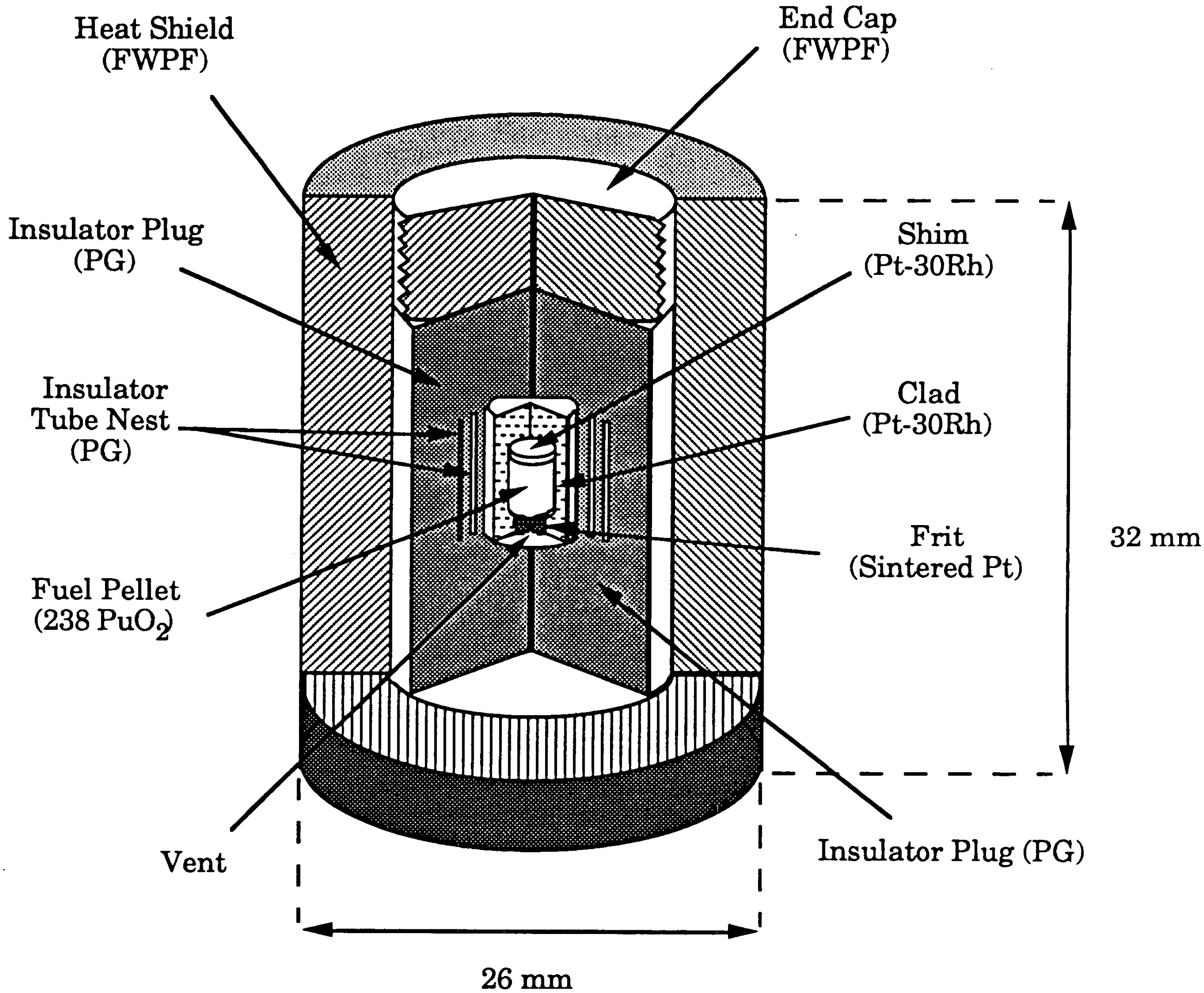



Figure 32: Light Weight Radioisotope Heater Unit

Approximately 320 LWRHU's are needed to produce the 352 Watts of heating necessary to keep the driller subsystems at the required temperature level of 283 K. With a height of 32 mm and diameter of 26 mm, the total volume and mass specifications are .42 $m^3$ and 12.8 kg respectively.

The thermal requirements of the drillers necessitate several devices to control their thermal environment. The devices chosen -- the thermal blanket, radioisotope heater unit, passive radiator, and Helium cooler -- combine to address those stringent requirements. The many approaches also provide a measure of flexibility and redundancy if one method does not work.

## 6.5 Communication

Transmission of information to and form the lunar driller will require a telecommunication system capable of performing telemetry, tracking and command functions. Telemetry involves transmission of scientific experiment, engineering systems status, and imaging systems data. Control of the driller, however minor it may be since the probe is almost fully robotic, will be performed by the telecommunication system's command function. This will make it possible to initiate any corrective commands if an unexpected problem should arise in the driller's operation. Lastly, the driller's position and velocity information will be made available through the system's tracking capability. This will be very important when the drillers make their descents to the chosen craters (Davis, 1989).

Advances in space communication and data transmission systems have made possible a number of options for communications with, and tracking of landers on the Moon. The equipment which can be employed ranges from radio frequency communication to the relatively new field of optical laser technology. Additionally, possible methods of transmission include direct communication from the Lunar surface or a number of signal relay options. Finally, the amount of communications coverage will also play a part in choosing the appropriate system. The following sections discuss the merits of the various data transmission equipment and methods (delineated below) as they pertain to this mission.

- Signal frequency
- Data relay versus direct links
- Communication coverage

### 6.5.1 Signal Frequencies

Communication with space vehicles requires reliable contact over continually expanding frequency ranges imposed by today's complex space systems. These requirements are best accommodated at higher frequencies, where extremely high-gain wide-bandwidth systems become more feasible (Krassner,1964). The communications equipment available can be categorized according to their operating frequency, and include S/X bands, Ka or MM-wave bands, and laser optical.

The operating frequency presently used, which benefits from a multitude of available equipment, is the S/X band. Most ground stations, including the backbone of the existing space communication system (the Deep Space Network), are based upon these frequencies. The S band is used for transmission to spacecraft and S/X bands are utilized for signal reception. Following (Table 1) is a description of the existing X-band's capability based on a proposed Manned Mars Mission. This mission employs orbiters and landers, as does the lunar polar drilling mission.

The second frequency option investigated was Ka/MM wave technology. This technology is currently not very advanced, however it is being pursued due to the benefits that could be enjoyed from its use. Ka/MM waves have the capability to support increased data rates, on the order of five to ten times, over the S/X band (White,1985). However, the lack of available equipment to support communication with the Moon over these frequencies, precludes their utilization at the current time.

Table 3: Return Link X-Band Capabilities*

| | RANGE | DATA RATE (MBPS) | EQUIV COLOR TV TRANSMISSION ** | | ANT DIA (M) | RF POWER (WATTS) |
|---|---|---|---|---|---|---|
| | | | FRAMES/ SEC. | SPATIAL RESOLUTION (PIXEL) | | |
| RELAY OPTION | | | | | | |
| ORBITERS | 1 AU | 9.2 | 30 | 640X480 | 4.88 | 230 |
| LANDERS | 33,000 KM | 10.0 | 32.5 | 640X480 | 1.31 | 20 |
| DIRECT OPTION | | | | | | |
| ORBITERS | 1 AU | 9.2 | 30 | 640X480 | 4.88 | 230 |
| LANDERS | 1 AU | 8.0 | 26 | 640X480 | 4.88 | 200 |

* A 2.3 dB adverse tolerance for cloud cover and 1.9 dB improvement in DSN is assumed
** A data compression equivalent to 1 bit per pixel is assumed

Optical frequency communication systems display the greatest benefits among the systems available. The benefits of lasers include: extremely high data rates, reduced power and equipment size, increased signal gain, and relative immunity to interception and jamming (White,1985). Conventional radio frequency systems have wavelengths from several centimeters to a few meters, while optical systems operate at wavelengths less than a few microns. Since signal divergence is proportional to the wavelength used, optical systems require smaller-diameter receivers than conventional radio frequency signals. By the same reasoning, a smaller and lower power telescope transmitter is required on the space vehicle. Use of an optical system would result in considerable savings in volume, mass, and power over conventional frequency systems.

A further benefit of optical systems is its possible use for navigation purposes. A laser's high modulation bandwidth allows short duration pulses to be generated easily. Optical instruments have the capability of providing accurate navigation information by means of these short pulses. The use of optical instruments combined with a precision pulse ranging system would be able to provide tracking information comparable to or better than that which can be attained with present day radio frequency navigation instruments (Chen,1988).

The application of optical laser communication to space missions requires certain developments in the component technologies which are now underway. One development involves the increase of laser power efficiency. Nd:YAG laser designs which achieve overall power efficiencies over 8.5% have been demonstrate, thus reducing the requirements for on-board electrical power generation. Additionally, the need to efficiently utilize photo energy is being addressed by advances in signal design and channel coding. Lastly, advances in semiconductor photodetector technology are also being made (Lesh,1987).

Lasers are an attractive communication option, however, the characteristics which make them so also make them hard to use. Currently the working laser systems can support transmission only up to an 8 km range. The drawbacks of this new technology are also its incompatibility with existing systems, atmospheric attenuation at earth, and a lack of proven optical systems. The benefits of the optical communication technology however, should not be overlooked. Assuming that by the launch date for this mission, sufficient advances will have been made in laser technology, laser communication equipment can be used to support medium range transmission in the vicinity of the moon. Through the use of lasers, the communication capabilities for this mission will be greatly enhanced. Presented in Table 4 is a comparison of the frequency options (White,1985).

Table 4: Communication Frequency Options

| | S/X | KA/MM | OPTICAL |
|---|---|---|---|
| BANDWIDTH | FEW MBPS | INCREASED | MUCH INCREASED |
| ANTENNA GAIN | REFERENCE | INCREASE OF 12 dB OVER X BAND | INCREASE OF 60-80 dB OVER Ka |
| IMMUNITY TO INTERCEPTION &JAMMING | POOR | BETTER | EXCELLENT |
| SIGNAL ACQUISITION | EASY | SATISFACTORY | DIFFICULT |
| POINTING ACCURACY | FEW ARC MIN ARC | ARC SEC RANGE SEC RANGE | ARC SEC TO SUB |
| LIFETIME | LONG | LONG | SHORT LASER LIFE |
| COMPATIBILITY WITH EXISTING SYSTEMS | YES | NO | NO |
| TECHNOLOGY STATUS | MATURE | IMMATURE (TECH. DEV. PLANNED) | IMMATURE (SOME RISK) |

Use of existing radio frequency methods with present technology would prove to be the best approach in devising reliable data transmission system. The reliability of the existing S/X band frequency transmission greatly outweighs the benefits of other options in long distance transmission.

### 6.5.2 Direct or Relay Transmission

The data transmission system options involve either direct communication to the Earth or the use of a relay system. The essential difference between the two approaches is at the lunar end of the communications system.

The direct communication method would transmit information directly from the lunar surface to Earth. This method of data transmission made possible nearly full time contact with the Earth during the Apollo missions and permitted several Earth stations to maintain continuous communication with a moon-based terminal (Krassner,1964). When considering this option, the advantages mentioned above appear to make this the desired mode of communication. However, local lunar signal obstructions such as mountains or crater rims and their polar location (Figure 33), combine to make direct line of sight communication improbable. In addition to the geometry problem, direct communication introduces the need for sophisticated equipment with the ability to transmit signals over large distances. This large transmitter-receiver distance requires both a larger antenna and powerful transmitter.

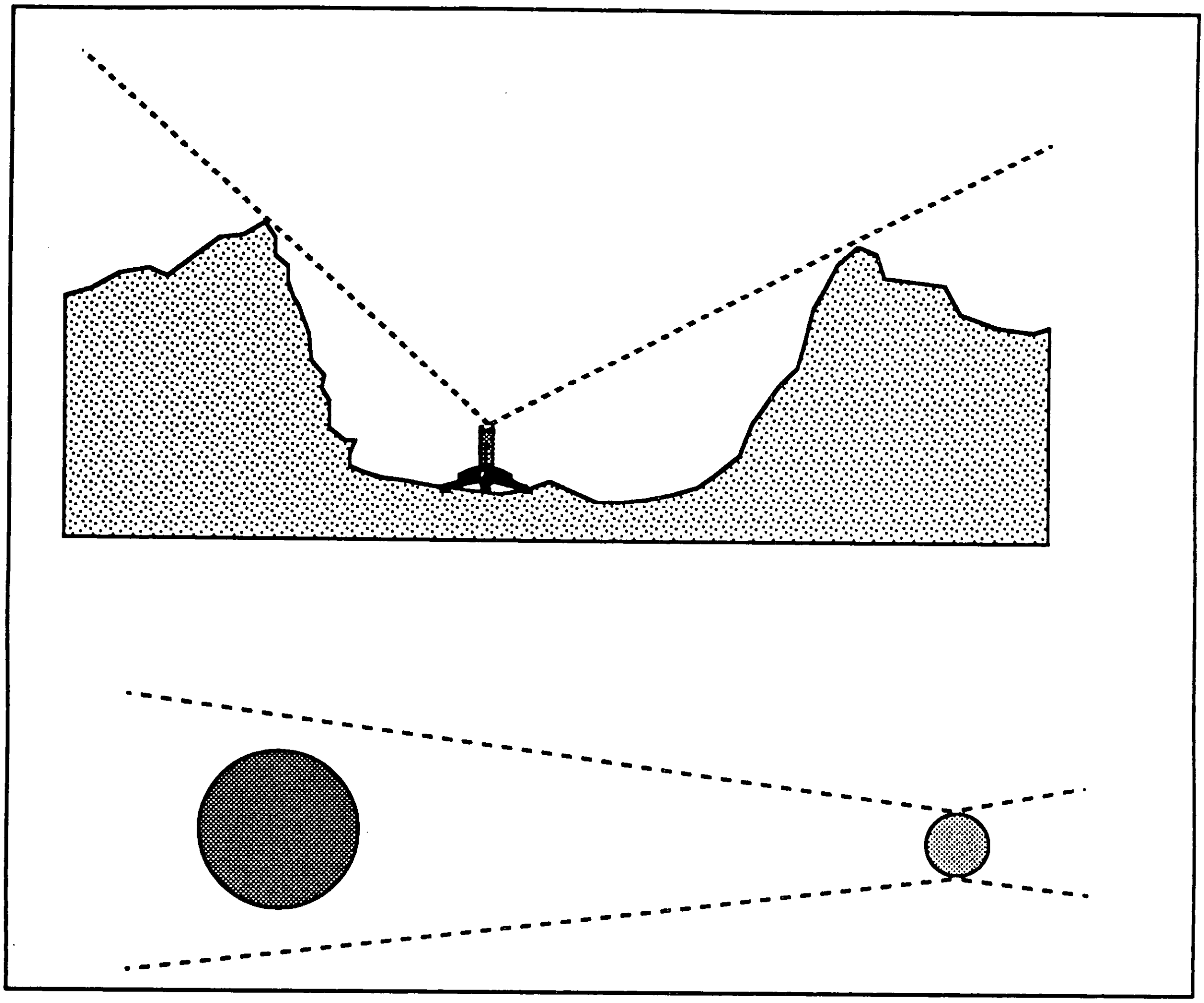


Figure 33: Exaggeration of Direct Communication Difficulties

In the relay system approach, the lunar orbiting remote sensing satellite or the manned lunar orbiter can be used as a data relay links between the Moon and Earth. The capabilities of the communications equipment aboard the orbiter would be far superior to lunar based equipment, since the orbiter is less constrained by mass limitations. The landers on the other hand, have to perform soft landings on the Moon. Therefore, the more equipment transported to the lunar surface, the more fuel necessary to soft land. It will not be possible to take full advantage of a relay orbiter, however, since lunar synchronous polar orbits are not possible. Therefore, the polar orbiter will loose contact with the driller during a portion of its orbit. This limitation results in the necessity for data storage equipment and repeater capability. The orbiter would require either a delayed (store-and-foreward) repeater or an instantaneous (real-time) repeater. Ideally it should have both so that when it is in view of the Earth, but not the landers, the delayed repeater could be used. When in view of both the landers and Earth, the instantaneous repeater may be used (Krassner, 1964). A further necessity for signal relay is the in lunar surface communication. Since the moon's horizon extends for only 5 km, lunar surface line-of-sight communication is impossible unless tower-mounted antennas are used. Therefore, the orbiters should also provide over-the-horizon communication between the manned lander and the drillers (Krassner,1964).

Regardless of the communication method chosen at the lunar end, signal reception at the Earth must also be considered. Either the Tracking and Data Relay Satellite System (TDRSS) and a ground receiving station, or the radio antenna Deep Space Network system (DSN) will be required for signal reception at Earth. The TDRSS satellites carry an omnidirectional antenna and six steerable antenna arrays for K-band, S-band and C-band communications. Since the antennas are directed toward Earth and the maximum a steerable one can be moved is +/- 30° in the North-South direction, a combination of the steerable and omnidirectional antenna is necessary. Additionally, a satellite receiving antenna on earth is necessary to receive the relayed signal (Walters,1987).

If the DSN approach to signal reception at Earth is followed, the communication link system will follow along the lines of the Apollo missions. Signals transmitted from the Moon will not require relay at the Earth, rather it will be received directly at the antenna sites. With this approach signal acquisition is possible from at least one of the network's antennas at any time. Since no costly modifications are necessary for reception by the DSN, and this reception system has been already been proven by the Apollo mission, it has been selected as the preferred option. Figure 34 presents the chosen relay communication approach.

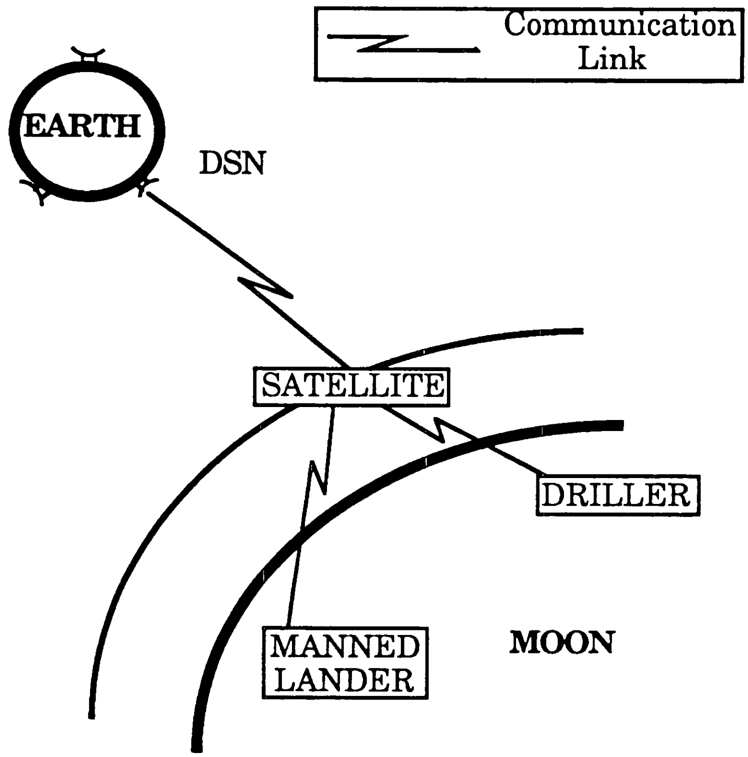



Figure 34: Communication Links

This coming decade will be marked by a reemergence of Earth remote sensing and space exploration missions. These missions, and those currently underway (Galileo, Voyager, Magellan), will require extensive use of both the TDRSS and DSN systems. In order not to impinge on this equipment vital for other missions, the lunar driller mission would ideally require a communication signal reception system of its own. Despite the short duration of this particular lunar mission, it will serve as a foundation for further lunar exploration missions which will require a dedicated communications system to provide full-time signal reception at Earth. Two approaches can be taken to solve these problems. The first possibility, and least expensive, is to modify a TDRSS satellite by attaching an outwardly pointing antenna to it. This antenna would exclusively serve lunar exploration operation, while the remaining earth-pointing antennas could serve terrestrial needs. The second option is to establish a ground antenna system similar to the DSN to be located adjacent to the current DSN antennas. These antenna facilities would serve lunar missions exclusively, thus not overloading the DSN system (Lundberg,1990).

### 6.5.3 Communication Coverage

The amount of communications coverage necessary for this mission is an influential factor in the communications system design. The precursor remote sensing satellite will serve as an information gathering orbiter and as a telecommunications relay orbiter, to serve the mission's communications needs. A number of orbits around the moon are possible each of which results in different communication times between the lunar vehicles and the satellite. The two options available are to utilize circular or elliptical orbits. Circular orbits offer a constant communication distance and uniform length coverage for all points on the moon. Elliptical orbits, on the other hand, offer the possibility of increasing the amount of coverage over a given point on the lunar surface. Communication to the orbiter becomes less reliable as its altitude at apoapsis increases. Thus, an orbit with a reasonable amount of communication time with the drillers, as well as a reasonably small altitude at apoapsis must be chosen.

Displayed below (Figure 35) is graph of various communication time and altitude at apoapsis options for elliptical orbits with a 110 km altitude at periapsis.

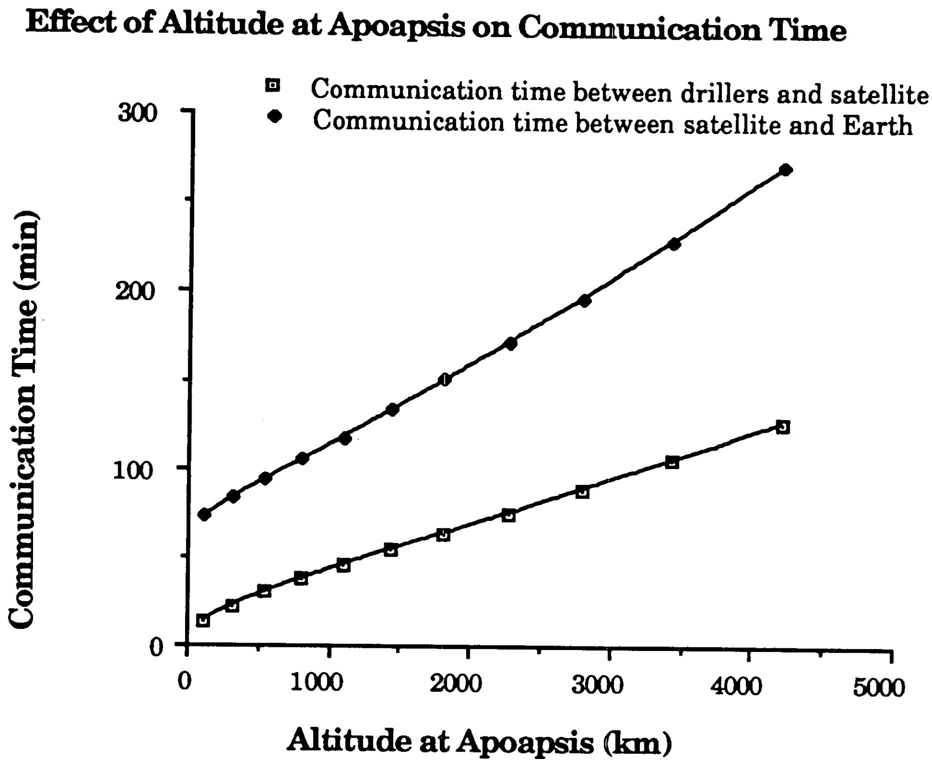
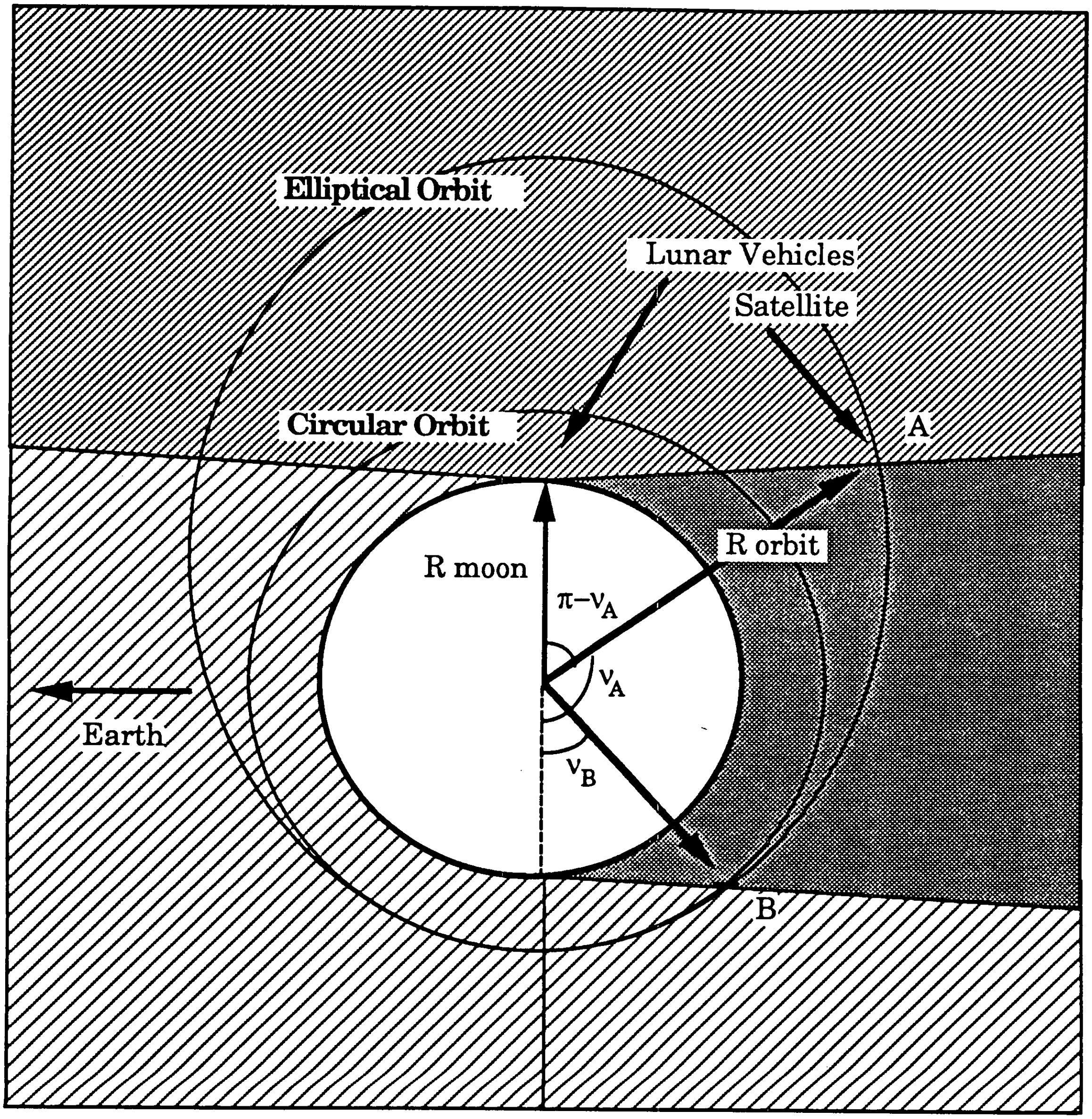



Figure 35 : Communication Times for Elliptical Satellite Orbits

An elliptical orbit with an altitude of 3700 km at apoapsis was chosen as adequate, and its eccentricity was determined to be .5 (see Appendix B). This orbit results in a orbit period of 5.5 hours, 1.9 hours of driller-orbiter communication, and 4.0 hours of possible earth-orbiter communication. Thus, for about one-third of the time the orbiter can communicate with the drillers. The 3.6 hour wait to regain communication is also acceptable, since it is not excessively long, and during that portion of the orbit the satellite can enter into delay-repeater mode. Figure 36 shows the communications coverage, for a



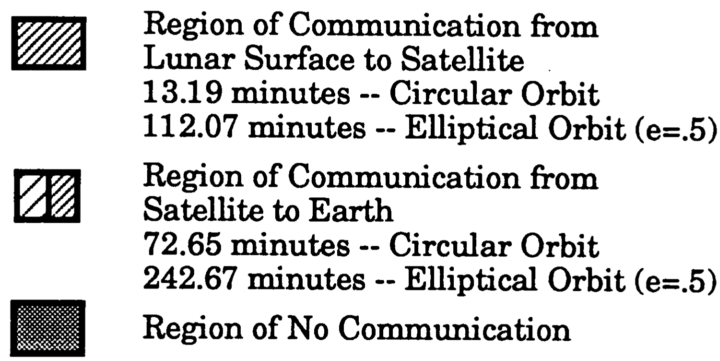




Figure 36: Communications Coverage

position at the Lunar pole, that could be provided via a 110 km circular orbit or the selected elliptical orbit.

Communication between the probe and the orbiter is possible only when the orbiter is above the horizon. During this portion of the orbit the satellite can relay real-time communication between the lunar surface and the Earth. Additionally, communication between the satellite and Earth is possible only when the orbiter is not on the Moon's far side. When this occurs and the orbiter is not in view of the lunar surface vehicles delayed repeater communication must be employed. It is possible to increase the coverage by using a second orbiter or a system of dedicated relay satellites. All of these options however introduce unnecessary amounts of additional hardware and mission costs.

The relay communication system appears to be the best communication link between the Lunar surface and Earth. It provides adequate real-time data relay transmission in addition to repeater relay transmission, without any sacrifice in mission cost. The direct approach, by contrast, would provide the possibility of full-time communication, but most likely will not be possible from the bottom of the polar craters.

Since many explorer spacecraft use technology inherited from previous spacecraft, the telecommunications system designed for this mission will be based on a spacecraft to be launched in the same time frame of this mission, namely the Cassini and Manned Mars missions (Davis,1989). The drillers S/X band relay communication equipment, using a polar orbiter relay satellite and DSN receivers at earth, will most likely have a mass of 5.0 kg (Beckman,1986). Since it will be using the standard NASA transponder, which produces an output power or 20 watts, less than 20 watts of power is required to run the equipment. Additionally, since the output power is generally fixed, the spacecraft antenna size will be dependent on the downlink frequency and receiver antenna size (Davis,1989).

Using the same frequency and DSN proposed by the Manned Mars mission, each driller will require a 1.3 meter diameter antenna (White,1985).

## 6.6 POWER SYSTEM

For space flight applications there are four distinct types of power systems: solar, fuel cell, nuclear and battery power systems. Solar power systems can be divided into solar photovoltaic and solar dynamic systems. Fuel cell systems are either non-regenerative and, currently under development, regenerative systems. Nuclear power systems are either a small-scale power supply from radioisotope-thermoelectric generators (RTG's) or large-scale, multi-megawatt nuclear reactors. Battery power systems store electrical energy in chemical form. These power systems will be detailed as they pertain to the mission at hand.

### 6.6.1 Solar Photovoltaic and Solar Dynamic

Solar Photovoltaic power generation is the process in which light energy from the sun is converted directly into electrical power through the use of an array of solar cells. Solar dynamic power systems use solar power to heat a working fluid to drive an engine to produce electricity. Since the mission will take place at the bottom of craters at the lunar poles, there will not be adequate light source to operate solar power systems. Therefore, both of these solar power systems are ruled out.

### 6.6.2 Nuclear Power

Radioisotope Thermoelectric Generators (RTGs) are used when the power requirement is less than 500 Watts. Power levels of 1 or 2 kilowatts could be obtained if the system was modified to a dynamic energy conversion system. In either case the power levels of RTG's is too low for the requirements of the lunar drilling mission. However, radioisotopes can be used for thermal generation. A lightweight radioisotope thermal generator produces 1.1 Watts of heat and has a mass of only 40 grams. Using radioisotopes for thermal heat generation would greatly reduce the power requirements of an electrical system.

Nuclear reactors, on the other hand, generate over 100 kilowatts of power. For a manned mission, the shielding requirements would be extensive for a nuclear reactor to operate safely. Also, the power levels produced is much higher than necessary for the lunar drilling mission. Therefore, nuclear reactor power generation will not be considered.

### 6.6.3 Fuel Cells

Fuel cells produce electrical power by the combination of a fuel with an oxidizer. The most common fuel cells use hydrogen/oxygen combination with water as a by-product. Non-regenerative fuel cells discard the water, or use it as a coolant for thermal management systems. In regenerative fuel cells the water is broken down into hydrogen and oxygen by electrolysis and reused; however, the electrolysis requires a secondary power source. The main concern in using fuel cells is the possibility of contaminating the core samples with the aqueous reactant by-product. Even if the reactive elements do not produce water, the fuel or oxidizer solutions are usually a molar concentration in water. For the lunar drilling mission, it would be necessary to find a fuel cell system that will not contain any water at all, yet still have high power output and low mass. However, as an example for sizing a water producing fuel cell, typical planning parameters are presented in Table 5. The Hydrogen / Oxygen fuel cell has a history of use due the large power output and low mass. The mass is low for for this cell because of the small molecular weights of Hydrogen and Oxygen.

Table 5: Hydrogen/Oxygen Fuel Cell (Non-Regenerative) Planning Parameters (Eagle,1988)

| | |
|---|---|
| Reactant Consumption Rate | 0.426 kg/kW-h |
| O/H Consumption Rate Ratio | 8:1 |
| Liquid Hydrogen Packaging Density | 59.3 kg/m$^3$ |
| Liquid Oxygen Packaging Density | 692 kg/m$^3$ |
| Liquid Hydrogen Tank Mass | 2.4 kg Tank/kg H |
| Liquid Oxygen Tank Mass | 0.25 kg Tank/kg O |

### 6.6.4 Batteries

Batteries are identified as either primary or secondary depending on their capability of being electrically recharged (Linden,1984). Primary cells are not capable of being recharged and are discarded after they have discharged. The primary cell is usually inexpensive, lightweight and portable. They generally have good shelf life, high energy density at low to moderate rates of discharge.

Secondary batteries, on the other hand, can be recharged after discharge to their original condition by passing current through them in the opposite direction to that of the discharge current. Secondary batteries are characterized by high power density, high discharge rate, flat discharge curves, and good low temperature performance (Linden,1984). Secondary batteries generally have lower energy densities and poorer charge retention than primary batteries, but they do have the advantage of being recharged several times.

A list of high energy density batteries is presented in Table 6 along with their type.

Table 6: Battery Power System Data

| Battery | Energy Density (Wh/kg) | Type |
|---|---|---|
| $Mg/MnO_2$ | 105 | Primary |
| Zn/HgO | 105 | Primary |
| $Li/SO_2$ | 280 | Primary |
| $Li/SOCl_2$ | 300 | Primary |
| $Al/O_2$* | 300 | Primary |
| Na/S* | 200 | Secondary |
| Zn/AgO | 90 | Secondary |

* indicates in development (System,1986)
Ref. (Linden,1984)

A battery of particular interest, which is currently under development, is the Aluminum / Oxygen battery. Although this is a primary battery, it has a very high energy density and can be used continuously as long as Aluminum is resupplied. A very promising secondary battery is the Sodium / Sulfur battery. This battery has a calender life of over ten years and a cycle life of 2500 but it has a charge depletion of less than one hour. Since solar photovoltaics and nuclear power has been ruled out due to various constraints, there will be no way to recharge a secondary battery after depletion.

### 6.6.5 Lunar Drill Power Requirements

Along with the drilling unit, the sub-systems which will require power include the computer, communication and sensor systems. Since the geology of the moon is relatively unknown, different materials are used to model the lunar surface layer. The materials used for modelling the drilling power requirements were pumice, unsorted cohesive conglomerate, vesicular basalt and dense basalt. Table 7 lists these materials along with the penetration rate and power consumption requirements for drilling a three meter hole. The values in Table 7 are based upon the research carried out for the development of the Apollo mission's three meter rotary percussion drill (Crouch,1965)

Table 7: Material Models for Lunar Drilling (Crouch,1965)

| Drilling Material | Penetration Rate (cm/min) | Watt hour/3m |
|---|---|---|
| Pumice | 300 | 8 |
| Unsorted Cohesive Conglomerate (micron to 0.5 cm particle size) | 150 | 16 |
| Vesicular Basalt (50% porosity) | 12 | 150-200 |
| Dense Basalt -1760 kg/cm$^3$ (25,000 psi) compressive strength | 3 | 900-1200 |

A linear relationship for the power requirement cannot be assumed due to the large amount of friction that will be present as the hole increases in length. It is more accurate to assume an exponential relationship, thus the actual power requirement for a fifteen meter hole will be close to ten times the power required for a three meter hole; however, the penetration rate will be assumed to remain constant.(Gray,1990)

Due to the uncertainty of the lunar geology, the worst case scenario will be used to assure an adequate power supply. It is known that the lunar poles are very mountainous, so it is an accurate assumption that the terrain will be very rocky. If dense basalt is used the power calculations, 12000 W-hrs will be required to drill a fifteen meter hole. Retrieval of the core is estimated to require no more than 75% of the power needed to make the hole. Therefore, the total power consumption to drill a fifteen meter hole and retrieve the core is approximately 21000 Watt-hours. Based on the data in Table 7, the rate of penetration for dense basalt leads to a drill time of approximately 8.3 hours. Approximately 7.7 hours should be sufficient to retrieve the core samples since the drill will no longer need to penetrate. Therefore, the total duration of the drilling operation is approximately 16 hours which leads to an average power requirement of 1230 Watts with a peak power requirement near 1450 Watts for the drilling unit. The maximum power requirement for the drilling unit and the other subsystems is presented in Table 8.

Table 8: Maximum Power Requirements for Lunar Driller

| Sub-system | Power(Watts) |
|---|---|
| Drilling Unit | 1450 |
| Computer | 30 |
| Communication | 20 |
| Guidance Sensors | 10 |
| Radar Altimeter | 28 |
| Camera | 36 |
| Total | 1574 Watts |

This total power requirement is for all sub-systems; however, the amount of time certain systems are active will vary. The break down of the active systems, approximate duration and power requirements of the sub-systems for each lunar drilling probe is presented in the following table.

| Phase / Sub-system | Power Requirement (Watts) | |
|---|---|---|
| <u>Landing Phase</u> (20 Minutes) | | |
| Computer | 30 | |
| Communication | 20 | |
| Guidance Sensors | 10 | |
| Radar Altimeter | 28 | |
| Camera | 36 | Total 124 |
| <u>Drilling Phase</u> (16 Hours) | | |
| Drilling Unit | 1450 | |
| Computer | 30 | |
| Communication | 20 | Total 1500 |
| <u>Drilling Shutdown</u> (1-5 Hours) | | |
| Computer | 30 | |
| Communication | 20 | Total 50 |
| Drill Stem Feed (40 Minutes) | | |
| 1/4 hp Motor | 186 | Total 186 |
| <u>Waiting Period</u> (1 Week to 3 Months) | | |
| Communication | μW | Total <1 |
| <u>Hop Maneuver</u> (15 Minutes) | | |
| Computer | 30 | |
| Communication | 20 | |
| Guidance Sensors | 10 | |
| Radar Altimeter | 28 | Total 88 |

Nuclear and Solar power generation options have been previously ruled out, thus only batteries and fuel cells require further analysis. Since a battery system will be unable to recharge, a primary type battery system must be used. The Aluminum / Oxygen battery system, which has an energy density rating of 300 W-hr/kg, would be very desirable if its development is completed by the target date. Otherwise, the battery of choice is the Lithium / Thionyl Chloride ($Li/SOCl_2$) battery or the Lithium / Sulfur Dioxide ($Li/SO_2$) battery which both have comparable energy densities (300 and 280 W-h/kg respectively).

Using 300 W-h/kg energy density batteries, the battery system for the lander and drill unit must be approximately 88 kg considering a three month waiting period or 83 kg for a one month waiting period. A small battery system for the hopper would be separate from the other power supply to conserve on weight for the hop maneuver. This battery system will be the power source for the Drill Stem Feed motors and the subsystems required for the hop maneuver. Thus, the batteries for the hopper unit would be approximately 0.5 kg.

A computer model was used to size a fuel cell capable of providing the suggested power requirements for the durations as listed above. This model is presented in Appendix B, Computer Models. Since the battery size for the hopper is small and easily set apart, it will remain the power source for the hopper. The fuel cell required to carry out the rest of the mission would be approximately 98 kg considering a three month waiting period and 96 kg for a one month waiting period. The problem with fuel cell designs is that they produce water as a by product or contain water in a molar solution; hence, it will be necessary to find a fuel cell which will not risk contamination of the core samples. Otherwise, extreme caution should be taken if an aqueous system is used.

Based on this analysis, a 300 W-h/kg battery system proves to be more economical than the fuel cell system since it is 13 kg less massive for the short waiting period scenario and 9 kg less massive for the long waiting period scenario. Also, by using only batteries as the power source, the problem of contaminating the core samples with extraneous water is eliminated. However, if batteries cannot remain charged for the duration of the mission, a non-water containing / producing fuel cell must be incorporated into the design.

### 6.7 Core Containment Unit Design

The core containment unit (the hopper) in the sample retrieval scenario is the sole point of contact of the lunar driller with humans once the mission has been initiated. After the coring is completed and the samples have been secured in the cannister, the hopper will be commanded to fly ("hop") to a designated landing site where the astronauts can retrieve the core samples. This scenario requires the hopper to operate in three modes:

(1) Fully automated. During the drilling and core extraction operation the hopper will be an integral part of the automated drilling sequence.

(2) Tele-operated. When the drilling is completed, the hopper will be commanded from an outside source to launch to a designated site.

(3) Manually-operated. When the hopper has landed, the astronauts must be able to easily retrieve the core samples.

This section will provide a detailed explanation of the equipment necessary to meet these requirements. Since the hopper must detach from the driller landing structure and fly to another site, it must carry its own motors and propellent, a guidance system, and landing system. The Oasis concept of this design is depicted in Figures 37 and 38. Minimizing the structural mass ratio and maximizing the structural strength to mass ratio have been accomplished through strategic use of composites and lightweight metallic alloys throughout the design. Figure 38 also shows where composites have been used to reduce the structural mass and identifies the nomenclature of the system components. All further descriptions of the hopper will use this nomenclature and in the "Drilling Operation" section which follows, the first mention of the components will be by both name and number in order to facilitate quick reference to the figure by the reader.

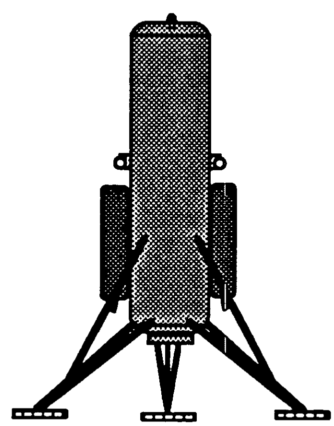
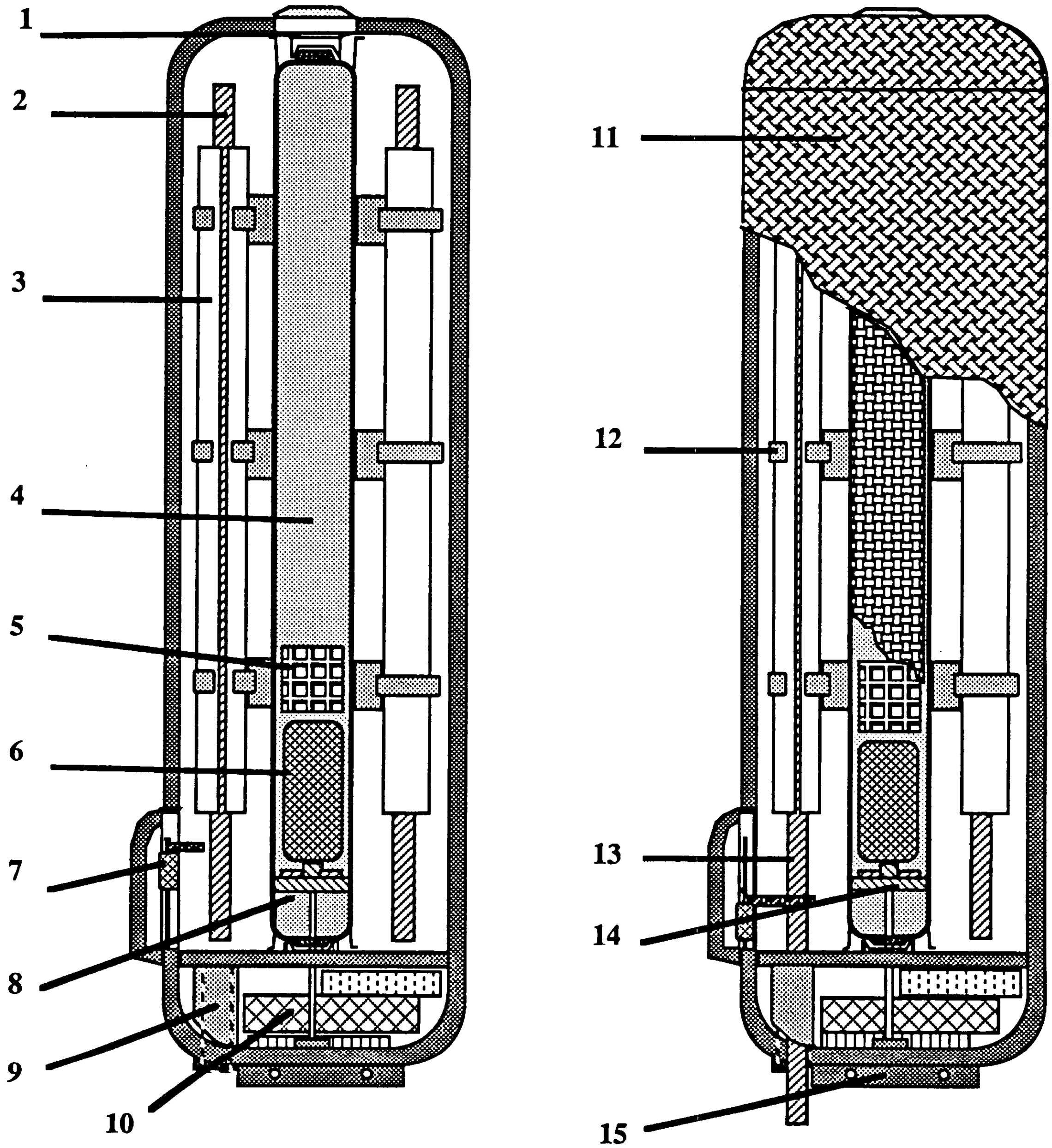


Figure 37: Hopper configuration with landing gear extended




1. UPPER BEARING AND SEAL
2. DRILL STEM (STOWED)
3. STORAGE RACK (COMP0SITE)
4. HUB SHAFT (COMPOSITE)
5. BATTERY
6. HUB DRIVE MOTOR
7. DRILL STEM TRANSFER ASSEMBLY
8. HUB DRIVE PLANETARY SET
9. DRILL STEM TRANSFER PORT
10. GUIDANCE/CONTROL AND COMMUNICATIONS
11. CANNISTER SHELL (COMPOSITE)
12. STORAGE CLAMP
13. DRILL STEM (TRANSFER POSITION)
14. STATIONARY SHAFT
15. HOPPER/LANDER INTERFACE CONNECTION

Figure : Drill core cannister cut-away

### 6.7.1 Drilling Operation

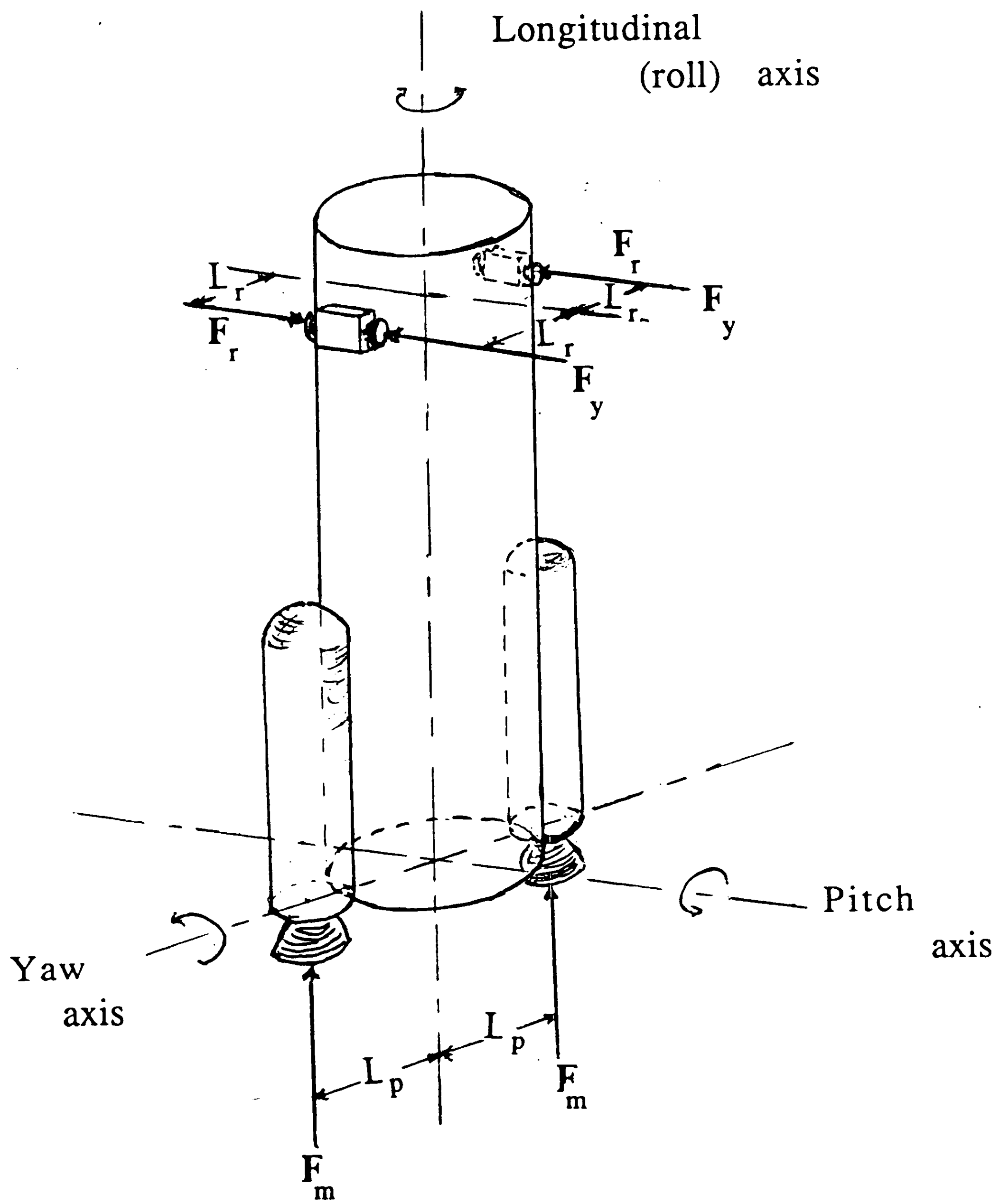
During the drilling operation, the hub shaft (4) is rotated by a planetary gear set (8) which is driven by the free-floating hub drive motor (6). This design allows the motor and its case to rotate on a stationary shaft and keeps the motor sealed away from the lunar dust. As the hub aligns an empty drill stem (2) over the stem transfer port (9), the vertical stem transfer drive assembly (7) grasps the base of the drill stem as the storage rack (3) simultaneously releases the stem. The vertical transfer drive assembly then lowers the stem through the transfer port into the drilling unit. The drilling unit will then maintain control of the stem during the drilling operation. The vertical transfer assembly then returns to its original position to wait until the next stem is required.

The extraction of the loaded core sample tube is simply the reverse of the insertion procedure. The drilling unit raises the core sample tube through the transfer port and into the cannister, returning the stem to the same storage rack from which it was originally removed. The core sample is then cut to the proper length by a cut-off saw located within the drilling unit. The transfer assembly then raises the sample into its fully stowed position in the storage rack, where it is clamped securely to await the launch out of the crater. Since it is desirable to minimize the structural mass ratio of a rocket propelled vehicle, composites were used. Mass reduction was also accomplished by designing the mechanisms within the hopper to operate as "slave" units. This implies that the hopper's operational commands during the drilling operations are received from a central processing unit located in the lander base. This also serves to simplify the programming of the operational command sequence by placing all of the programming into this central processing unit.

### 6.7.2 Retrieval Flight

When the coring is completed and a sample retrieval landing site has been selected and prepared by the astronauts, the hopper is ready to detach itself from the lander and fly to the designated landing site. The operational details of this guidance system is explained in the "Hopper Navigation" section. Attitude control is achieved through four roll/yaw thrusters and the main propulsion motors. Since the main propulsion motors are mounted symmetrically about the vertical axis (which has been designated the longitudinal, or roll axis), the line of intersection passing through the motors has been designated as the yaw axis. The designation of the control axes is shown in Figure 39. By using differentially pulsed thrust, the main motors can provide pitch control. This arrangement minimizes the number of thrusters needed to provide three-axis attitude control for the hopper, thus further reducing the structural mass ratio of the hopper. The roll/yaw thruster pairs have been placed above the center of mass and in the plane of the roll and yaw axes. Prior to launch, the hopper/lander connection and umbilical are separated either by servo-mechanical locks or by use of explosive bolts. During the flight, the landing legs are extended and locked. When all of the hoppers have landed at their selected sites, the astronauts can then retrieve the core samples.

# ATTITUDE CONTROL




| | |
|---|---|
| $F_m \times L_p$ = Pitch moment | $F_y \times L_r$ = Rolling moment |
| $F_r \times L_r$ = Yaw moment | |

Figure 39: Hopper control axis designation

### 6.7.3 Core Retreival

In order to facilitate retrieval of the core samples, an access panel was designed into the side of the containment cannister. The fasteners attaching this panel are similar to the quick-release fasteners which are prevalent in the access panels of aircraft. These fasteners may be released by the astronauts using a speed handle equipped with a special bit. Speed handles are easy to use even with gloved hands, and will make panel removal quick and easy. After the panel is removed, the hub must be rotated and the storage racks opened in order to retrieve the core samples. The operation of the hub and storage racks will be controlled by switches located just inside the access hatch. The storage rack which is lined up with the access hatch will be opened by the astronauts via the control switch so that they can retrieve the core sample. The open rack is then closed so that it will not rub against the cannister, and the hub is rotated to expose the next storage rack. When all the core segments have been retrieved, sealed, and labeled, the astronauts may proceed to the next hopper. This sequence of operations is repeated at each of the hopper units until all of the core samples have been retrieved.

## 6.8 Guidance Control and Navigation

Guidance systems will be required on the probe for two phases of the mission. One system will be needed for landing the probe within the crater; And a second system will be required for guiding the hopper to the retrieval sight. The purpose of the system during each maneuver will be to measure the probes altitude and velocity. Therefore, the central components of these guidance systems, will be an altimeter and a velocity sensor. The velocity and altitude data obtained by these instruments will be used to determine the rocket operations necessary for steering the probe to its destination and controlling its descent. The following sections discuss the operation of the systems selected for the descent and the hop maneuver.

### 6.8.1 Lander Descent

The guidance system for landing the probe in the craters will be modelled after the surveyor system, the characteristics of which are shown in Table 5. This system will consist of two radar devices, the first of which is an altitude marking radar (AMR). The AMR is a pulse transmitting range measuring radar (Skolnik,1970) which will be preset for a mark altitude of 60 nmi. When the system observes a time difference, from pulse transmission to echo reception, which corresponds to this altitude, a signal will be transmitted to the probe computers to initiate the landing sequence, i.e. fire the retro-rocket. The AMR will be placed inside the nozzle of the retro-rocket so that once this initiation signal is transmitted and the retro-rocket ignited, the pressure of the retro exhaust gas will expel the AMR from the nozzle ( Skolnik,1970).

Table 9: Surveyor Guidance System Characteristics.(Skolnik,1980)

| | |
|---|---|
| Frequency | Ku Band (12.5- 18 GHz) |
| Operating Altitude | |
| Velocity Sensor | Below 15,000 m |
| Altimeter | Below 13,000 m |
| Accuracy | |
| Velocity Sensor | 3.0 percent |
| Altimeter | 5.0 Percent |
| Modulation | |
| Velocity Sensor | Continuous Wave |
| Altimeter | FM-CW |

Below 13,000 meters, the descent will be controlled via information provided by the second radar device, the radar altimeter and doppler velocity sensor (RADVS). The RADVS system is a frequency modulated - continuous wave radar (Skolnik,1970). Continuous wave is used for velocity determination (Hovanessian,1984). The wave is transmitted at a known frequency; However, due to the velocity of the probe, the echo is received at a different frequency. By measuring this frequency change, the velocity of the probe can be calculated. The problem with continuous wave radar is that if the signal is uniform, there is no means of determining when a received portion of the signal was transmitted; and without this information range can not be determined (Skolnik,1980). This dilemma is solved by varying the frequency of the transmission signal. This frequency modulation provides timing marks within the signal (Hovanessian, 1984). This enables the system to determine the transmission time of a given portion of the received wave. Range can then be determined by measuring the time between the transmission of the modulation to the reception of its echo. Therefore, as the probe descends, the computer will use velocity and range data supplied by a continuous wave-frequency modulation radar system as reference for descent operations.

### 6.8.2 Sight Selection

One characteristic absent from, and unnecessary for, the surveyor system was the ability to discriminate between landing sights. The rocky terrain in the polar craters, however, mandates a system which can steer the probe, away from large rocks and surface irregularities, to a suitable landing position. A suitable landing sight is distinguished by a surface height differential of less than 1.8 meters. If the probe lands on an area with a greater differential, the legs will not be able to conform to the terrain and keep the probe in an upright position,as illustrated in Figure 40. In accordance with this limitation, the altimeter system used for sight selection must be able to resolve height differentials of a single decimeter.

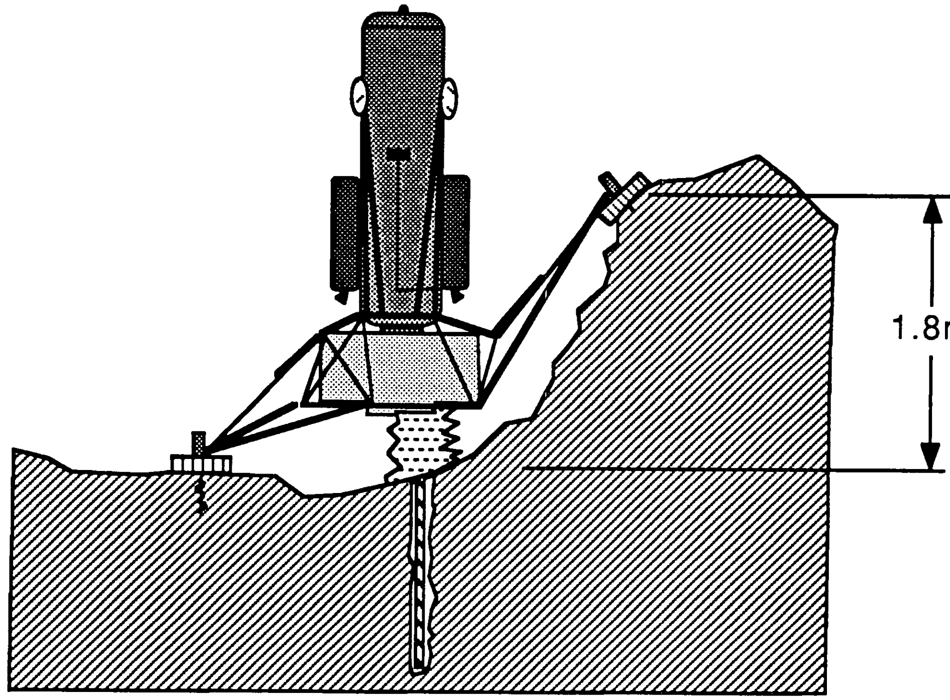



Figure 40 : Landers maximum leg excursion for a vertical landing

In order to achieve sight selectiveness, Oasis is proposing a differential Radar system which will operate during the quasi-hover portion of the probes descent. This system will consist of three transmitting / receiving antennas, one attached to the end of each of a landing legs, as shown in Figure 41. A unique pulse will be transmitted simultaneously form each of the antenna. The relative times at which the pulses are received will be used to determine the relative altitudes of the legs. These altitude differentials will be transmitted to the probe's computer. If all three differentials are less than 1.8 meters, then the probe will continue its vertical descent. If, however, one or more of the differentials exceed the 1.8 meter limitation, a signal will be sent to the vernier motors to steer the probe away from the largest differentials Figure 41 illustrates the steering process where R1 designates the highest terrain, and R2 the lowest. Additionally, the magnitude of the 'steer' maneuver will be proportional to the magnitude of the maximum differential. By controlling the magnitudes of the maneuvers in this manner, the adjustments made by the probe will decrease as the differentials decrease and near the maximum acceptable value. Eventually the system will lock in on an acceptable landing position. When all of the altitude differentials are less than 1.8 m, the probe will descend vertically to touchdown.

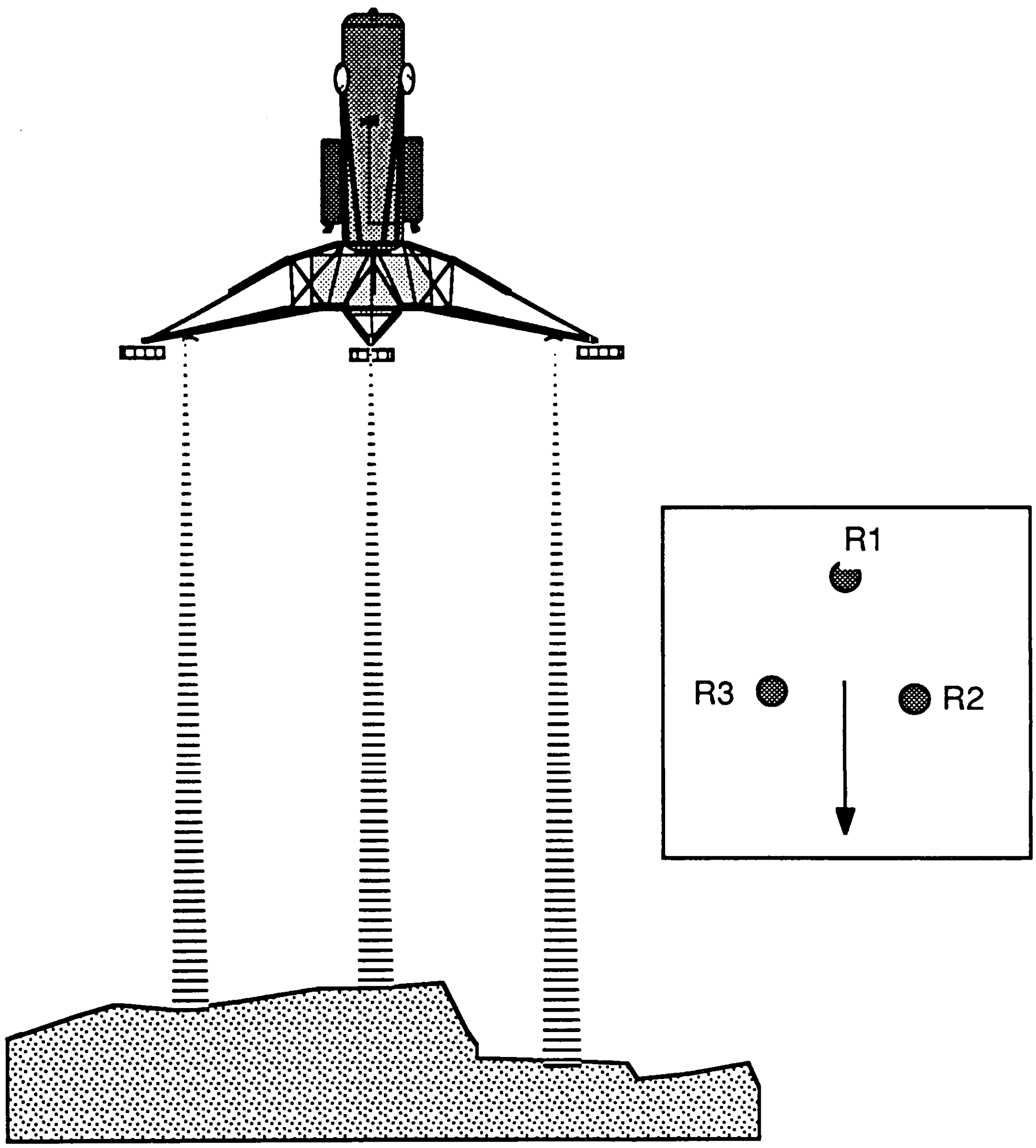



Figure 41: Differential Radar altimeter and Steering operation.

### 6.8.3 Hopper Guidance

When the astronauts arrive on the moon to retrieve the core samples, they will place three radio beacons in the retrieval region. A signal will, then, be broadcast to each of the probes, via the command module or communication satellite, commanding them to launch towards the retrieval sight. Each of the probes will be instructed to land at a unique set of coordinates within the retrieval area.

During the initial portion of the hop, the lip of the crater will block the signal from the beacons as illustrated in Figure 42. Therefore, during this time period, the hoppers will be guided by a signal from the command module or communication satellite. As the hoppers rise out of the craters they will lock in on the signal being transmitted from the radio beacons.

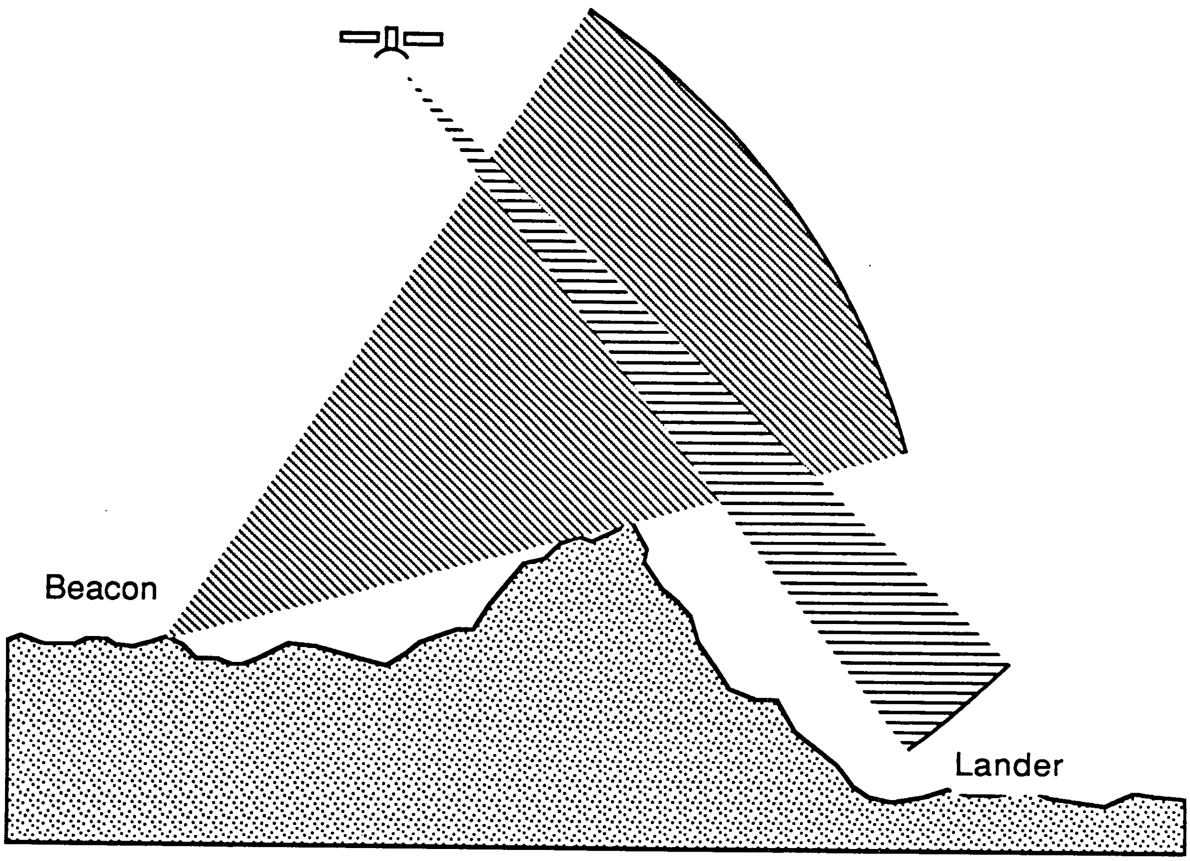


Figure 42: Signal blockage due to crater lip.

Each of these beacons will transmit a continuous wave signal at a unique frequency. The hopper will use these three signals to determine its location and velocity. The hopper's computer will then make the trajectory corrections necessary to bring the probe to a landing at its prescribe coordinates. This guidance method, known as triangulation, will be used to guide each of the hoppers to predetermined retrieval coordinates, where the astronauts can extract the core samples.

### 6.8.4 Location and Velocity Determination

The system will determine velocity by observing the doppler shifts in the received signals. The magnitude of the hopper's velocity with respect to each of the beacons will be calculated from these shifts. The computer, however, will not be able to determine the hopper's velocity vector until it has the directions of these velocities.This requires knowledge of the position of the hopper.

In order for the probe to determine its position from observation of the signals, it must be able to measure the time between the signals' transmission and reception. Frequency modulation is, therefore, used to provide timing marks for the recognition of transmission and reception times (Hovanessian,1984). At regular synchronized time intervals, frequency variations will be simultaneously broadcast from each of the beacons. The relative times at which these variations are received will then be used to determine the relative distance to each of the beacons. These three relative ranges will define a line of position along which the hopper is located (Lundberg,1990). In order to identify where along this line the hopper is located, one absolute range must be determined. This will be accomplished by including a additional timing mark within the broadcast code of one of the beacons (Lundberg,1990). A frequency modulation, different from the modulations used for determining the relative positions, will be broadcast at a predetermined time and predetermined intervals thereafter. The transmission times of this modulation will thereby be specified. The absolute range will then calculated from the time delay between reception and transmission. This absolute range, used in conjunction with the three relative ranges, will determine the position of the hopper. With the position determined, the velocity vector can be resolved. Then, The position and velocity of the hopper will be used to determine the necessary correction for guiding the hopper to its destination.

### 6.8.5 Mass Breakdown

Based on the design sections for the lunar coring probe, the following mass breakdown is presented in Table 6. This breakdown is a key component in the determination of the amount of fuel required for the successful deployment of the coring probe.

Table 10: Mass Breakdown of the Lunar Coring Probe

| | Lunar Orbit | After Main Retro Ejec. | Lunar Surface |
|---|---|---|---|
| **Subsystems** | kg | kg | kg |
| Battery | 83 | 83 | 83 |
| Radar | 20 | 20 | 20 |
| Thermal Control | 5 | 5 | 5 |
| Communications | 19 | 19 | 19 |
| Structure | 23 | 23 | 23 |
| Drilling Equipment | 90 | 90 | 90 |
| **Hopper** | | | |
| Hopper (empty&dry) | 40 | 40 | 40 |
| Propellant | 36 | 36 | 36 |
| **Propulsion** | | | |
| Main Retro | 286 | | |
| Vernier Retro | 34 | 34 | |
| **Total** | 636 | 350 | 316 |

# 7.0 Conclusion

A preliminary design has been performed for a robotic lunar lander. It by no means represents the only design solution to the request for proposal, but it is believed to offer the best compromise to the various mission demands. The flexibility of the design of the lander offers mission planners the option of choosing among the mission scenarios presented in this report, or any variation of them.

As NASA heads toward the Moon again, a polar mission in search of water is considered to be essential. Oasis hopes that the design presented in this document will give engineers a useful starting point in designing such a mission.

# 8.0 Enabling Technologies

An objective of the lunar drill design is to incorporate existing "off the shelf" technology in order decrease the time required for actual deployment; however, it turned out that certain technologies must be developed. The following sections detail some of the advances which need to be made in order to allow the full design to be implemented.

## 8.1 Coring Techniques

Since the objective of the lunar coring mission is to determine if water exists on the moon, investigations must be conducted to verify the feasibility of dry coring. On earth, large amounts of water are used to lubricate and cool the drill string. For the lunar drill mission, dry lubricants such as carbon or Teflon must be used as a lubricant to avoid contaminating the core samples.

Another area of development of drilling techniques is fully automated remote coring. The landing probes will be completely autonomous, thus there will be very little or no control from ground crews. The onboard computer will have to monitor the drill status and make all adjustments as required. If anything should go wrong, the computer should be flexible to adapt to the situation and correct the error or modify the situation.

## 8.2 Terrain Adaptable Landing Gear

The drilling probe incorporates a new concept to deal with the unknown terrain at the bottom of polar craters. The flexibility incorporated in the legs allows the lander to touch down on very uneven terrain while maintaining a nearly vertical orientation of the drilling mechanisms. Since this type of landing gear has never been used before, research must be carried out to assure the reliability of such a system. Also, the algorithm developed to monitor the landing process may need to be expanded upon in order to deal with unforeseen situations.

## 8.3 Differential Radar Steering

To assist in landing the drilling probe in the polar craters, a differential radar steering system is proposed. The differential radar will determine the variations of terrain levels and compensate by steering the probe away from the greatest terrain variation. The system would require a transmitter and receiver on each leg capable of high accuracy range measurements at a high rate. The range differentials must have a tolerance of one tenth meter for the computer to determine if landing is possible at the selected site or if it is necessary to steer away.

## 8.4 Laser Communication System

A laser communication system has been developed for atmospheric applications of less than 10 miles. It has not, however, been proven for long range space applications. A laser communication system has the advantages of very accurate data transfer at a very high rate with little distortion. The disadvantages are that the laser communication system is presently quite massive and has a large power requirement. If laser communication technology advances in time for it to be a feasible option, its benefits to the mission will be considerable.

## 8.5 Power System

A large amount of power is required for the autonomous drilling probe. To reduce the mass of the probe either a fuel cell system or a high energy density batteries must be used. The fuel cell systems studied either produce water as a by-product of the combination of fuel and oxidizer, or the fuel or oxidizer are contained in a molar solution of water. If a fuel cell system is used, it is imperative that a nonaqueous cell be developed to avoid the possibility of core sample contamination.

High energy density batteries such as Lithium / Sulfur Dioxide and Lithium / Thionyl Chloride batteries have been developed but they have not been proven in space. An Aluminum / Oxygen battery, which is currently under development and has been suggested for use on the moon, has the possibility of being resupplied with Aluminum for continuous use. If the suggested battery systems prove to be adequate for the mission purposes, the risk of core sample contamination would be eliminated.

Although these proposed technology advancements are not imperative for the successful completion of the lunar drilling mission, they would contribute to the overall quality of the design. Since very little is known about the lunar pole craters and many limitations are imposed by this site selection, the above enabling technologies have been suggested as a means with dealing with these problems.

# 9.0 Operations Management

Oasis Lunar Systems has a company structure which allows for the flexibility necessary for the development and flow of ideas. Presented below is the company's management approach, a schedule of the project's milestones, directories of the company's branches, and a company roster.

## 9.1 Management Approach

The management approach of Oasis is to channel the talents of each employee into areas of the company where they will be most useful. The company is divided into a technical and an administrative section with the first section headed by the CEO and the second by the company Vice-President. The technical area is comprised of the Orbits, Spacecraft, and Science divisions while the administrative area is made up of Management, Documentation, and Graphics.

## 9.2 Schedule

The project was be undertaken with the aim of fulfilling the requirements with the highest efficiency while also producing a quality design. During the allotted time for completion of the driller design a few schedule changes were necessary. A schedule of the milestone dates encountered during the project is presented below. Additionally, the project's task timeline is presented in Figure 43.

| | | |
|---|---|---|
| March 7, 1990 | -- | Preliminary Design Review Presentation |
| March 26, 1990 | -- | Preliminary Design Review Report |
| April 8, 1990 | -- | Project Abstract |
| April 20, 1990 | -- | NASA/USRA Presentation |
| April 25, 1990 | -- | Final Presentation |
| May 4, 1990 | -- | Final Report |

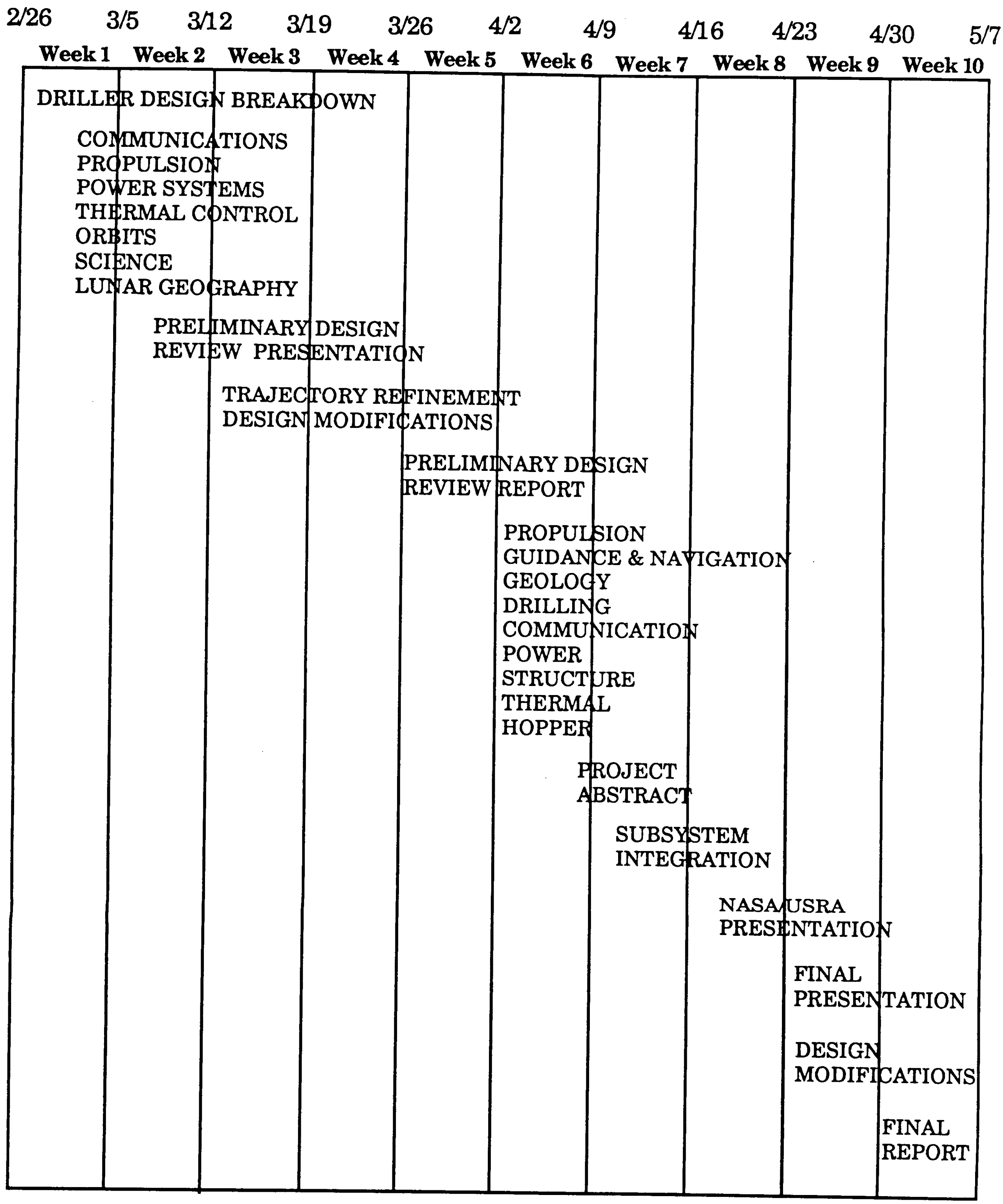



Figure 43 Task Timeline

## 9.3 Management Structure

The company is comprised of technical and administrative sections with three divisions each. In designing the driller, the technical section was reorganized to make development of the driller's components an easier task. The driller design was broken down into the general areas of driller subsystems, drilling operations, and hopper design. Organizational layouts of the company's engineering and administrative branches are presented below in Figures 44 and 45 respectively. They are followed by a company roster shown in Table 9. Note that every company member serves in the engineering capacity.

Figure 44: Engineering Directorate

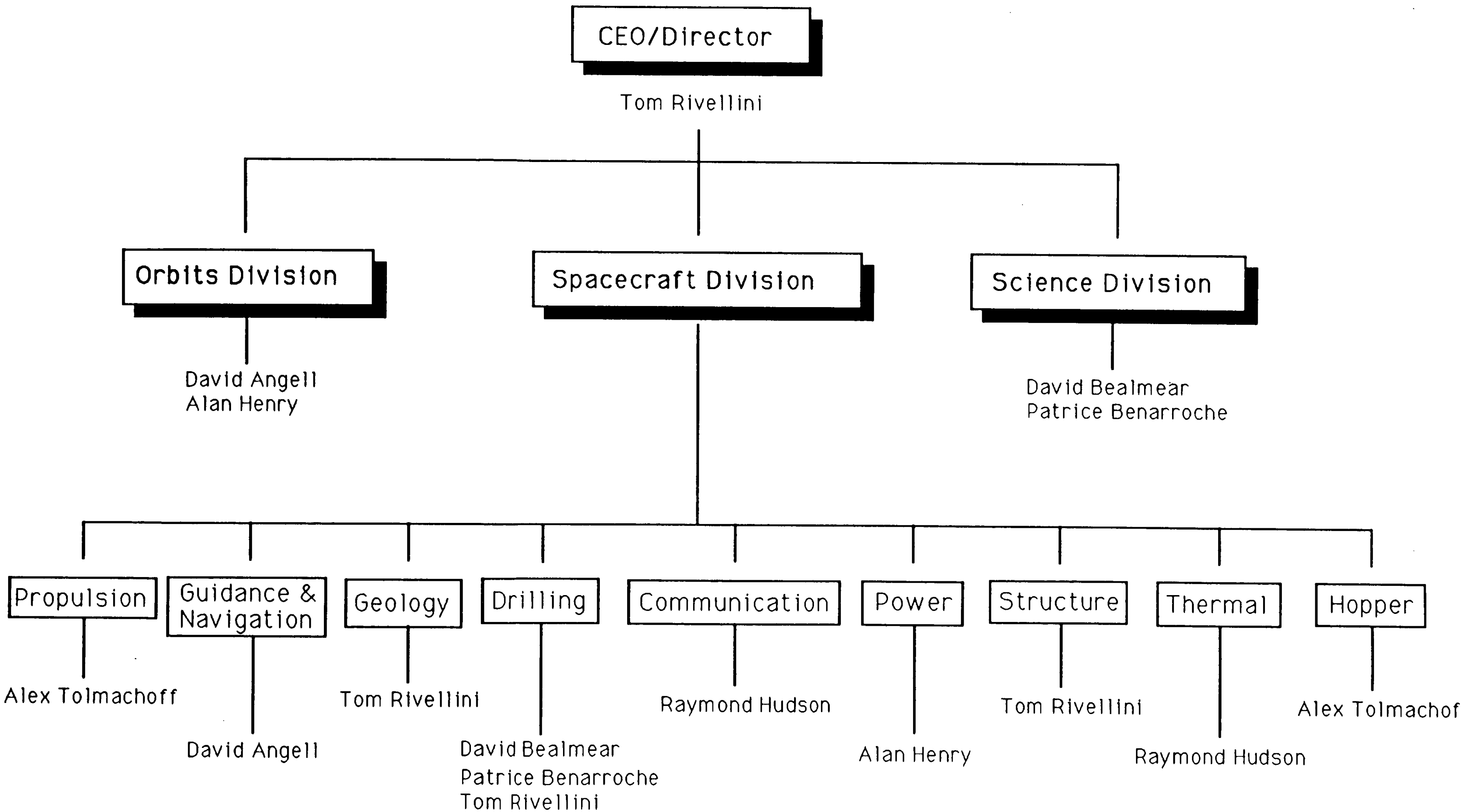


Figure 45: Administrative Directorate

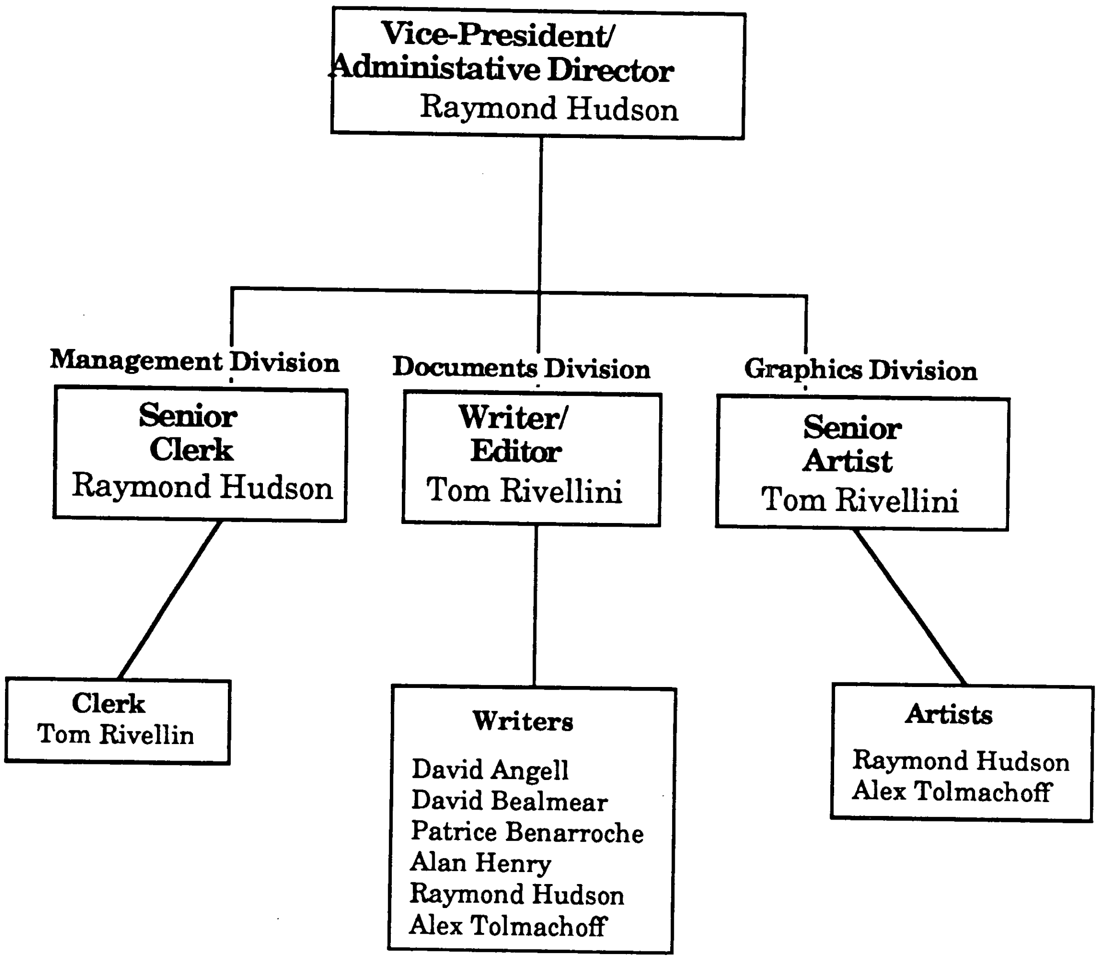


Table 11: Company Roster

| Name | Permanent Address | Telephone |
|---|---|---|
| David Angell | 2309 Bengal Lane<br>Plano, TX 75023 | (214) 596-6888 |
| David Bealmear | 7227 Hwy 290<br>Austin, TX 78723 | (512) 929-3384 |
| Patrice Benarroche | 8 Rue De L'Esteron<br>06510 Carros, France | (011-33) 93-29-02-32 |
| Alan Henry | 6419 Los Altos<br>El Paso, TX 79912 | (915) 584-6698 |
| Raymond Hudson | P.O Box 75<br>Nederland, TX 77627 | (409) 722-7390 |
| Tom Rivellini | 1624 Flagstone Court<br>Clearwater, FL 34616 | (813) 442-8289 |
| Alex Tolmachoff | 905 Bodark Lane<br>Austin, TX 78745 | (512) 445-4437 |

## 9.4 Cost Analysis

The total project cost for the lunar driller design work was approximately \$18,180.00. This section introduces the accounting methods utilized in addition to the costs accrued through its implementation.

### 9.4.1 Cost Accounting Approach

The budget for the project is based upon practicality and common sense, rather than any sophisticated economic analysis model. The task of designing such a mission requires a large number of individuals and resources. The resources necessary for the most part are available at no cost except for computers, and personnel -- which require the largest monetary outlay. The personnel our company has available performed several tasks and fulfilled a number of rolls. The weekly man-hours spent in each position were arrived at by averaging the number of hours spent in those positions throughout the design investigation.

July 24, 1990:

Pages 129 and 130 removed because of funding information.

PHILIP N. FRENCH
Document Evaluator

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# Appendix A Trajectory Analysis

## Orbital Operations

The object of this section is to present the necessary trajectory and energy considerations for the mission. This section is divided into the criteria, options, and lunar trajectory. The trajectory options and preliminary velocity change requirements, calculated in order to size engines and determine fuel requirements, are outlined.

### Criteria

In all manned missions, the first and foremost consideration is safety of the crew. For this reason, an elliptical transfer orbit will be analyzed so that in case of a miss fire or booster shut-down the space capsule will have the capability of returning to Earth. The underlining criteria for trajectory optimization is cost. Due to the high expense of space launches, it is desirable to minimize the fuel required; however, the time of flight becomes an important design criterion for a manned mission. Another important factor is the launch windows, which are currently being studied.

### Options

This mission is unique compared to previous lunar missions due to the need to achieve lunar polar orbit. Numerous non-coplanar options exist for reaching polar orbit. Non-coplanar trajectories, however, involve plane changes which are expensive. Due to the expense, plane changes are undesirable and in fact they are unnecessary. By targeting a point in space just below or just above the lunar poles and injecting into translunar orbit along the appropriate launch asymptote, the trajectory is reduced to a coplanar problem. The coplanar trajectory problem has already been solved for the Apollo missions. The Apollo delta V's will therefore be used to approximate the delta V's for this mission, except for the unmanned translunar trajectory in which a Hohmann transfer will be used to minimize the delta V's

## Lunar Trajectory

For the preliminary delta V calculations, patched conic approximations will be used. The conic sections include a low Earth circular orbit, an elliptical transfer orbit to the moon, an elliptical lunar insertion, and a low lunar circular orbit. Other delta V calculations involve moon landing and moon launch into low lunar orbit.

The initial low Earth orbit is assumed to be 160 nautical miles or 296.32 km. The circular velocity of this orbit is 7.73 km/sec. The insertion speed for a 50.2 hour transfer to the moon is approximately 10.9 km/sec. Thus, the initial delta V is approximately 3.17 km/sec from circular orbit. For the unmanned portions of the mission, a Hohmann transfer will be used and the initial delta V will be 3.10 km/sec.

Upon arrival to the lunar sphere of influence, the moon will be traveling faster than the spacecraft. The moon travels at 1.018 km/sec and the spacecraft velocity is approximately 0.199 km/sec. The relative velocity vector of the spacecraft with respect to the moon is 0.819 km/sec. The spacecraft will be drawn toward the moon and at nearest approach a Hohmann transfer will be used to put the craft into a low lunar orbit at an altitude of 60 nautical miles. The delta V required to reduce altitude and then stabilize into circular orbit is approximately 0.858 km/sec.

To de-orbit the landing vehicle the delta V required is approximately 0.0254 km/sec based on a Hohmann transfer. Due to the mountainous lunar polar terrain however, Hohmann transfers cannot be used for lunar surface landing trajectories. The Apollo de-orbit delta V of 0.0649 km/sec is therefore used to approximate this change. Then, to slow the lander down for a soft landing, a delta V of approximately 1.842 km/sec is required. A delta V of this same magnitude is required for the ascent into LLO and finally, a delta V of 0.67 km/sec is necessary to escape the moon on a trans-Earth trajectory. . Summing the delta V's and adding 15 percent for maneuvering and Earth injection gives an approximate total delta V of 9.7 km/sec.

# Appendix B Computer Models

## Computer models for the fully robotic missions.

The variables and rules for TK! Solver models are given on the following pages. These models were used to estimate the propellent and structural masses and propellent volumes for each stage of the fully robotic missions.

### FULLY ROBOTIC SAMPLE RETURN

VARIABLES

| St Input | Name | Output | Unit | Comment |
|---|---|---|---|---|
| ?1.623119 | gm | | m/s^2 | gravity of moon |
| 15000 | h | | m | maximum vertical height of hopper |
| 150000 | x | | m | horizontal distance of hopper |
| 1.7227573 | linmass | | kg/m | linear mass of core sample |
| 10 | lcor | | m | length of core sample |
| | Mhpay | 1717.5721 | kg | mass of hopper with core samples |
| 250 | Mls | | kg | mass of landing structure |
| 1704 | Vdo | | m/s | delta-v to de-orbit |
| 0 | VL90 | | m/s | plane change deltav to polar |
| 1000 | VLOI | | m/s | deltav at LOI |
| 3200 | VTLI | | m/s | deltav at TLI |
| 1 | N | | | # of drillers |
| 0 | Msat | | kg | mass of comsat |
| .2770083 | PCTFh | | | % fuel of hopper |
| .2770083 | PCTFL | | | % fuel of main retro motors |
| .166666 | PCTFo | | | % fuel of TLI booster |
| 870 | rhoMMH | | kg/m^3 | density of MMH |
| 1430 | rhoNTO | | kg/m^3 | density of NTO |
| 71 | rhoLH | | kg/m^3 | density of LH2 |
| 1185 | rhoLOX | | kg/m^3 | density of LO2 |
| | Vh | 220.66622 | m/s | vertical deltav of hopper |
| | Vx | 551.66556 | m/s | horizontal deltav of hopper |
| | Vt | 3500 | m/s | total deltav of hopper |
| | Mph | 4738.26 | kg | propellent mass of hopper |
| | Mpl | 563.36675 | kg | propellent mass of landing/hover stage |
| | Mt | 7781.7894 | kg | total mass of driller in lunar orbit |
| | MLOI | 10047.133 | kg | total mass at LOI |

| | | | | |
|---|---|---|---|---|
| | MTLI | 23476.034 | kg | total mass at TLI |
| | FuVolh | 1.5086636 | m^3 | fuel volume of hopper |
| | OXVolh | 2.3956102 | m^3 | oxidizer volume of hopper |
| | FuVoll | .17937617 | m^3 | fuel volume of landing/hover stage |
| | OXVoll | .28483181 | m^3 | oxidizer volume of landing/hover |
| | FuVolo | 28.657378 | m^3 | fuel volume of TLI booster |
| | OXVolo | 8.5851629 | m^3 | oxidizer volume of TLI booster |
| | Mpo | 12208.092 | kg | propellent mass of TLI booster |
| | MLpo | 7781.7894 | kg | mass of all drillers and satellite in |
| | Mhop | 6912.086 | kg | mass of hopper minus core samples |
| 220.7 | Vhov | | m/s | delta-v required to come to hover posi |
| | Mt2 | 14737.871 | kg | total mass in lunar orbit before first |
| .2770083 | PCTFr | | | % fuel of landing/hover stage |
| | FuVolr | 5.2551611 | m^3 | fuel volume of main retro |
| | OxVolr | | kg | oxidizer volume of main retro |
| | Mpret | 6323.7104 | kg | mass of main retro propellent |
| | Mp2 | 2059.4032 | kg | propellent mass of LOI burn |
| | Mlpay | 7162.086 | kg | mass of lander delivered to lunar surf |

# RULES

## S Rule

```
"Rule set 1:  Defining the total delta v required as a function
"          of x and h
"
"
t= sqrt(2*h/gm)
Vh= sqrt(2*gm*h)
Vx= x/(2*t)
Vt= 3500
"
"Rule set 2:  Computing total mass inserted into lunar polar orbit
""
Mhpay= 1700 + 1.02*(linmass*lcor)
Vsp= exp(Vt/(309.9*9.81))
Mph= Mhpay*(1 - Vsp)/(0.1*Vsp - 1.1)
Mhop= (Mhpay - 1.02*(linmass*lcor)) + 1.1*Mph
Vspl= exp(Vhov/(309.9*9.81))
Mlpay= Mhop + Mls
Mpl= Mlpay*(1- Vspl)/(0.1*Vspl - 1.1) + 20
Mt= Mlpay + 1.1*Mpl
Vret= exp(Vdo/(309.9*9.81))
Mpret= Mt*(1 - Vret)/(0.1*Vret - 1.1)
Mt2= Mt + 1.1*Mpret
"
"Rule set 3:  Computing mass at LOI and TLI
"
Vsp2= exp(VLOI/(444.4*9.81))
Vsp1= exp(VTLI/(444.4*9.81))
MLpo= (N*Mt) + Msat
Mp2= (MLpo)*(1 - Vsp2)/(0.1*Vsp2 - 1.1)
Mpo= (MLpo + 1.1*(Mp2))*(1 - Vsp1)/(0.1*Vsp1 -1.1)
MLOI= MLpo + 1.1*(Mp2)
MTLI= MLOI + 1.1*Mpo
"
"Rule set 4:  Computing propellent tank volumes
"
"
PCTOXh= 1-  PCTFh
PCTOXl= 1 - PCTFL
PCTOXo= 1 - PCTFo
PCTOXr= 1 - PCTFr
FuVolh= PCTFh*(Mph/rhoMMH)
OXVolh= PCTOXh*(Mph/rhoNTO)
FuVoll= PCTFL*(Mpl/rhoMMH)
OXVoll= PCTOXl*(Mpl/rhoNTO)
FuVolo= PCTFo*(Mpo/rhoLH)
OXVolo= PCTOXo*(Mpo/rhoLOX)
FuVolr= PCTOXr*(Mpret/rhoMMH)
OXVolr= PCTOXr*(Mpret/rhoLOX)
```

# FULLY ROBOTIC DATA RETURN

## VARIABLES

| St Input | Name | Output | Unit | Comment |
|---|---|---|---|---|
| ?1.623119 | gm | | m/s^2 | gravity of moon |
| 15000 | h | | m | maximum vertical height of hopper |
| 150000 | x | | m | horizontal distance of hopper |
| 1.7227573 | linmass | | kg/m | linear mass of core sample |
| 10 | lcor | | m | length of core sample |
| 100 | Mhpay | | kg | mass of hopper with core samples |
| 250 | Mls | | kg | mass of landing structure |
| 1704 | Vdo | | m/s | delta-v to de-orbit |
| 0 | VL90 | | m/s | plane change deltav to polar |
| 1000 | VLOI | | m/s | deltav at LOI |
| 3200 | VTLI | | m/s | deltav at TLI |
| 1 | N | | | # of drillers |
| 0 | Msat | | kg | mass of comsat |
| .3090625 | PCTFh | | | % fuel of hopper |
| .3984063 | PCTFL | | | % fuel of main retro motors |
| .166666 | PCTFo | | | % fuel of TLI booster |
| 870 | rhoMMH | | kg/m^3 | density of MMH |
| 1430 | rhoNTO | | kg/m^3 | density of NTO |
| 71 | rhoLH | | kg/m^3 | density of LH2 |
| 1185 | rhoLOX | | kg/m^3 | density of LO2 |
| | Vh | 220.66622 | m/s | vertical deltav of hopper |
| | Vx | 551.66556 | m/s | horizontal deltav of hopper |
| | Vt | 0 | m/s | total deltav of hopper |
| | Mph | 0 | kg | propellent mass of hopper |
| | Mpl | 39.802646 | kg | propellent mass of landing/hover stage |
| | Mt | 376.21079 | kg | total mass of driller in lunar orbit |
| | MLOI | 485.72887 | kg | total mass at LOI |
| | MTLI | 1134.9494 | kg | total mass at TLI |
| | FuVolh | 0 | m^3 | fuel volume of hopper |
| | OXVolh | 0 | m^3 | oxidizer volume of hopper |
| | FuVoll | .01822715 | m^3 | fuel volume of landing/hover stage |
| | OXVoll | .01674477 | m^3 | oxidizer volume of landing/hover stag |
| | FuVolo | 1.3854416 | m^3 | fuel volume of TLI booster |
| | OXVolo | .4150499 | m^3 | oxidizer volume of TLI booster |
| | Mpo | 590.21048 | kg | propellent mass of TLI booster |
| | MLpo | 376.21079 | kg | mass of all drillers and satellite in |
| | Mhop | 82.427876 | kg | mass of hopper minus core samples |
| 220.7 | Vhov | | m/s | delta-v required to come to hover posi |
| | Mt2 | 738.73747 | kg | total mass in lunar orbit before first |
| .322581 | PCTFr | | | % fuel of landing/hover stage |
| | FuVolr | .25661699 | m^3 | fuel volume of main retro |
| | OxVolr | | kg | oxidizer volume of main retro |
| | Mpret | 329.56971 | kg | mass of main retro propellent |
| | Mp2 | 99.561896 | kg | propellent mass of LOI burn |
| | Mlpay | 332.42788 | kg | mass of lander delivered to lunar surf |

## RULES

```
"Rule set 1:  Defining the total delta v required as a function
"          of x and h
"
"
t= sqrt(2*h/gm)
Vh= sqrt(2*gm*h)
Vx= x/(2*t)
Vt= 0
"
"Rule set 2:  Computing total mass inserted into lunar polar orbit
""
Mhpay= 1700 + 1.02*(linmass*lcor)
Vsp= exp(Vt/(309.9*9.81))
Mph= Mhpay*(1 - Vsp)/(0.1*Vsp - 1.1)
Mhop= (Mhpay - 1.02*(linmass*lcor)) + 1.1*Mph
Vspl= exp(Vhov/(273*9.81))
Mlpay= Mhop + Mls
Mpl= Mlpay*(1- Vspl)/(0.1*Vspl - 1.1) + 11.0
Mt= Mlpay + 1.1*Mpl
Vret= exp(Vdo/(294*9.81))
Mpret= Mt*(1 - Vret)/(0.1*Vret - 1.1)
Mt2= Mt + 1.1*Mpret
"
"Rule set 3:  Computing mass at LOI and TLI
"
Vsp2= exp(VLOI/(444.4*9.81))
Vsp1= exp(VTLI/(444.4*9.81))
MLpo= (N*Mt) + Msat
Mp2= (MLpo)*(1 - Vsp2)/(0.1*Vsp2 - 1.1)
Mpo= (MLpo + 1.1*(Mp2))*(1 - Vsp1)/(0.1*Vsp1 -1.1)
MLOI= MLpo + 1.1*(Mp2)
MTLI= MLOI + 1.1*Mpo
"
"Rule set 4:  Computing propellent tank volumes
"
"
PCTOXh= 1- PCTFh
PCTOXl= 1 - PCTFL
PCTOXo= 1 - PCTFo
PCTOXr= 1 - PCTFr
FuVolh= PCTFh*(Mph/rhoMMH)
OXVolh= PCTOXh*(Mph/rhoNTO)
FuVoll= PCTFL*(Mpl/rhoMMH)
OXVoll= PCTOXl*(Mpl/rhoNTO)
FuVolo= PCTFo*(Mpo/rhoLH)
OXVolo= PCTOXo*(Mpo/rhoLOX)
FuVolr= PCTOXr*(Mpret/rhoMMH)
OXVolr= PCTOXr*(Mpret/rhoLOX)
```

## Computer models for the manned/robotic missions.

The variables and rules for TK! Solver models are given on the following pages. These models were used to estimate the propellent and structural masses and propellent volumes for each stage of the orbital rendezvous and landing missions respectively. Provision has also been made to make these computations for a single launch mission which sends more than one drill unit.

# MANNED/ROBOTIC ORBITAL RENDEZVOUS

## VARIABLES

| St | Input | Name | Output | Unit | Comment |
|---|---|---|---|---|---|
| | 1.623119 | gm | | m/s^2 | gravity of moon |
| | 15000 | h | | m | maximum vertical height of hopper |
| | 150000 | x | | m | horizontal distance of hopper |
| | 1.7227573 | linmass | | kg/m | linear mass of core sample |
| | 15 | lcor | | m | length of core sample |
| | | Mhpay | 56.309324 | kg | mass of hopper with core samples |
| | 250 | Mls | | kg | mass of landing structure |
| | 1704 | Vdo | | m/s | delta-v to de-orbit |
| | 0 | VL90 | | m/s | plane change deltav to polar |
| | 1000 | VLOI | | m/s | deltav at LOI |
| | 3200 | VTLI | | m/s | deltav at TLI |
| | 1 | N | | | # of drillers |
| | 0 | Msat | | kg | mass of comsat |
| | .390625 | PCTFh | | % | fuel of hopper |
| | .3984063 | PCPCTFo | | % | fuel of TLI booster |
| | 870 | rhoMMH | | kg/m^3 | density of MMH |
| | 1430 | rhoNTO | | kg/m^3 | density of NTO |
| | 71 | rhoLH | | kg/m^3 | density of LH2 |
| | 1185 | rhoLOX | | kg/m^3 | density of LO2 |
| | | Vh | 220.6622 | m/s | vertical deltav of hopper |
| | | Vx | 551.66556 | m/s | horizontal deltav of hopper |
| | | Vt | 1704 | m/s | total deltav of hopper |
| | | Mph | 120.93688 | kg | propellent mass of hopper |
| | | Mpl | 361.86548 | kg | propellent mass of landing/hover |
| | | Mt | 811.12912 | kg | total mass of driller in lunar orbit |
| | | MLOI | 1047.2555 | kg | total mass at LOI |
| | | MTLI | 2447.0072 | kg | total mass at TLI |
| | | FuVolh | .05429996 | m^3 | fuel volume of hopper |
| | | OXVolh | .0515356 | m^3 | oxidizer volume of hopper |
| | | FuVoll | 16571205 | m^3 | fuel volume of landing/hover stage |
| | | OXVoll | .15223496 | m^3 | oxidizer volume of landing/hover |
| | | FuVolo | 2.9870808 | m^3 | fuel volume of TLI booster |
| | | OXVolo | 89486816 | m^3 | oxidizer volume of TLI booster |
| | | Mpo | 1272.5015 | kg | propellent mass of TLI booster |
| | | MLpo | 811.12912 | kg | mass of all drillers and satellite in |
| | | Mhop | 163.07709 | kg | mass of hopper minus core samples |
| | 220.7 | Vhov | | m/s | delta-v require to come to hover |

| | | | |
|---|---|---|---|
| | Mt2 | kg | total mass in lunar orbit before first |
| .32258 | PCTFr | | % fuel of landing/hover stage |
| | FuVolr | m^3 | fuel volume of main retro |
| | OxVolr | m^3 | oxidizer volume of main retro |
| | Mpret | kg | mass of main retro propellent |
| | Mp2 170.94734 | kg | propellent mass of LOI burn |
| | Mlpay 328.95883 | kg | mass of lander on the lunar surface |

## RULES

```
S Rule
 "Rule set 1:  Defining the total delta v required as a function
 "         of x and h
 "
 "
 t= sqrt(2*h/gm)
 Vh= sqrt(2*gm*h)
 Vx= x/(2*t)
 Vt= sqrt(Vh^2 + Vx^2)
 "
 "Rule set 2:  Computing total mass inserted into lunar polar orbit
 ""
 Mhpay= 38.7372 + 1.02*(linmass*lcor)
 Vsp= exp(Vt/(273*9.81))
 Mph= 2.2*Mhpay*(1 - Vsp)/(0.1*Vsp - 1.1)
 Mhop= (Mhpay - 26.2628) + 1.1*Mph
 Vspl= exp(Vdo/(294*9.81))
 Mlpay= Mhop + Mls
 Mpl= Mlpay*(1- Vspl)/(0.1*Vspl - 1.1)
 Mt= Mlpay + 1.1*Mpl
 "
 "Rule set 3:  Computing mass at LOI and TLI
 "
 "
 Vsp3= exp(VL90/(444.4*9.81))
 Vsp2= exp(VLOI/(444.4*9.81))
 Vsp1= exp(VTLI/(444.4*9.81))
 MLpo= (N*Mt) + Msat
 Mp3= MLpo*(1 - Vsp3)/(0.1*Vsp3 - 1.1)
 Mp2= (MLpo + 1.1*Mp3)*(1 - Vsp2)/(0.1*Vsp2 - 1.1)
 Mpo= (MLpo + 1.1*(Mp3 + Mp2))*(1 - Vsp1)/(0.1*Vsp1 -1.1)
 MLOI= MLpo + 1.1*(Mp3 + Mp2)
 MTLI= MLOI + 1.1*Mpo
 "
 "Rule set 4:  Computing propellent tank volumes
 "
 "
 PCTOXh= 1-  PCTFh
 PCTOXl= 1 - PCTFL
 PCTOXo= 1 - PCTFo
```

```
FuVolh= PCTFh*(Mph/rhoMMH)
OXVolh= PCTOXh*(Mph/rhoNTO)
FuVoll= PCTFL*(Mpl/rhoMMH)
OXVoll= PCTOXl*(Mpl/rhoNTO)
FuVolo= PCTFo*(Mpo/rhoLH)
OXVolo=PCTOXo*(Mpo/rhoLOX)
```

# MANNED/ROBOTIC LANDING MISSION

## VARIABLES

| St | Input | Name | Output | Unit | Comment |
|---|---|---|---|---|---|
| | 1.623119 | gm | | m/s^2 | gravity of moon |
| | 15000 | h | | m | maximum vertical height of hopper |
| | 150000 | x | | m | horizontal distance of hopper |
| | 1.7227573 | linmass | | kg/m | linear mass of core sample |
| | 15 | lcor | | m | length of core sample |
| | | Mhpay | 65.095387 | kg | mass of hopper with core samples |
| | 250 | Mls | | kg | mass of landing structure |
| | 1704 | Vdo | | m/s | delta-v to de-orbit |
| | 0 | VL90 | | m/s | plane change deltav to polar |
| | 1000 | VLOI | | m/s | deltav at LOI |
| | 3200 | VTLI | | m/s | deltav at TLI |
| | 1 | N | | | # of drillers |
| | 0 | Msat | | kg | mass of comsat |
| | .390625 | PCTFh | | | % fuel of hopper |
| | .3984063 | PCTFL | | | % fuel of main retro motors |
| | .166666 | PCTFo | | | % fuel of TLI booster |
| | 870 | rhoMMH | | kg/m^3 | density of MMH |
| | 1430 | rhoNTO | | kg/m^3 | density of NTO |
| | 71 | rhoLH | | kg/m^3 | density of LH2 |
| | 1185 | rhoLOX | | kg/m^3 | density of LO2 |
| | | Vh | 220.66622 | m/s | vertical deltav of hopper |
| | | Vx | 551.66556 | m/s | horizontal deltav of hopper |
| | | Vt | 594.16199 | m/s | total deltav of hopper |
| | | Mph | 36.478407 | kg | propellent mass of hopper |
| | | Mpl | 288.17586 | kg | propellent mass of landing/hover |
| | | Mt | 645.95228 | kg | total mass of driller in lunar orbit |
| | | MLOI | 833.99436 | kg | total mass at LOI |
| | | MTLI | 1948.7032 | kg | total mass at TLI |
| | | FuVolh | .01637859 | m^3 | fuel volume of hopper |
| | | OXVolh | 01554478 | m^3 | oxidizer volume of hopper |
| | | FuVoll | .13196676 | m^3 | fuel volume of landing/hover stage |
| | | OXVoll | .12123411 | m^3 | oxidizer volume of landing/hover |
| | | FuVolo | 2.3787972 | m^3 | fuel volume of TLI booster |
| | | OXVolo | .71263886 | m^3 | oxidizer volume of TLI booster |
| | | Mpo | 1013.3717 | kg | propellent mass of TLI booster |
| | | MLpo | 645.95228 | kg | mass of all drillers and satellite in |
| | | Mhop | 78.958834 | kg | mass of hopper minus core samples |
| 220.7 | | Vhov | | m/s | delta-v require to come to hover pos |
| | | Mt2 | | kg | total mass in lunar orbit before first |
| .32258 | | PCTFr | | | % fuel of landing/hover stage |
| | | FuVolr | | m^3 | fuel volume of main retro |
| | | OxVolr | | m^3 | oxidizer volume of main retro |
| | | Mpret | | kg | mass of main retro propellent |
| | | Mp2 | 170.94734 | kg | propellent mass of LOI burn |
| | | Mlpay | 328.95883 | kg | mass of lander on the lunar surface |

RULES

```
S Rule
"Rule set 1:  Defining the total delta v required as a function
"          of x and h
"
"
t= sqrt(2*h/gm)
Vh= sqrt(2*gm*h)
Vx= x/(2*t)
Vt= sqrt(Vh^2 + Vx^2)
"
"Rule set 2:  Computing total mass inserted into lunar polar orbit
""
Mhpay= 38.7372 + 1.02*(linmass*lcor)
Vsp= exp(Vt/(273*9.81))
Mph= 2.2*Mhpay*(1 - Vsp)/(0.1*Vsp - 1.1)
Mhop= (Mhpay - 26.2628) + 1.1*Mph
Vspl= exp(Vdo/(294*9.81))
Mlpay= Mhop + Mls
Mpl= Mlpay*(1- Vspl)/(0.1*Vspl - 1.1)
Mt= Mlpay + 1.1*Mpl
"
"Rule set 3:  Computing mass at LOI and TLI
"
"
Vsp3= exp(VL90/(444.4*9.81))
Vsp2= exp(VLOI/(444.4*9.81))
Vsp1= exp(VTLI/(444.4*9.81))
MLpo= (N*Mt) + Msat
Mp3= MLpo*(1 - Vsp3)/(0.1*Vsp3 - 1.1)
Mp2= (MLpo + 1.1*Mp3)*(1 - Vsp2)/(0.1*Vsp2 - 1.1)
Mpo= (MLpo + 1.1*(Mp3 + Mp2))*(1 - Vsp1)/(0.1*Vsp1 -1.1)
MLOI= MLpo + 1.1*(Mp3 + Mp2)
MTLI= MLOI + 1.1*Mpo
"
"Rule set 4:  Computing propellent tank volumes
"
"
PCTOXh= 1-  PCTFh
PCTOXl= 1 - PCTFL
PCTOXo= 1 - PCTFo
FuVolh= PCTFh*(Mph/rhoMMH)
OXVolh= PCTOXh*(Mph/rhoNTO)
FuVoll= PCTFL*(Mpl/rhoMMH)
OXVoll= PCTOXl*(Mpl/rhoNTO)
FuVolo= PCTFo*(Mpo/rhoLH)
OXVolo= PCTOXo*(Mpo/rhoLOX)
```

## Fuel Cell Sizing Model

This model was written on TK! Solver to size fuel cells given the power requirement and duration of activation.

```
(Rule Sheet)
"This model calculates the system parameters for a fuel cell system
 "To run the model, first enter the power required (P) in the desired
 "units.  Next, enter the cycle time (tc) in the desired units.
 "The model is now ready to run.
 "
 "
* mfc= specm*p
* mes= 0.70*specm*p
* mfo= 1.05*cr*p*tc  "includes 5% extra to account for small leaks and boil-off
* MolH2= mfo/(wH2+wO2/2)
* mH2= MolH2*wH2
* mO2= mfo-mH2
* mH2O=mfo
* mH2tank= 0.62*mH2
* mO2tank= 0.19*mO2
* mH2Ot= .07*mH2O
* mtotal= mfc+ mes+ mfo+ mH2tank+ mO2tank+ mH2Ot
* ptherm= (1-eff)*p/eff
```

```
(Variable Sheet)
   28.8        specm        kg/kWe       Specific mass of power system
   .055        p            kWe          Power required
   .426        cr           kg/(kWe-h    Consumption rate of fuel and oxidizer
L  168         tc           hr           Cycle time
   .00202      wH2          kg/mol       Mass of 1 mole of H2
   .032        wO2          kg/mol       Mass of 1 mole of O2
   .5          eff                       Efficiency of power system
               mfc     1.584     kg      Mass of fuel cell subsystem
               mes     1.1088    kg      Mass of electrolysis subsystem
L              mfo     4.133052  kg      Total mass of fuel and oxidizer
               MolH2   229.35916         Moles of hydrogen required
L              mH2     .4633055  kg      Mass of hydrogen required
L              mO2     3.6697465 kg      Mass of oxygen required
               mH2O    4.133052  kg      Mass of water produced
               mH2tank .28724941 kg      Mass of hydrogen storage tank
               mO2tank .69725184 kg      Mass of oxygen storage tank
               mH2Ot   .2893136  kg      Mass of water storage tank
L              mtotal  8.0996669 kg      Total mass of fuel cell power system
               ptherm  .055      kW      Thermal power to radiate
```

**Communication Time Calculations**

## Constants

$R_{moon}$=9.3849352E2 n.mi.
$R_{apoapsis}$=9.3849352E2 n.mi. + 60 n.mi.=9.99849352E2 n.mi.
$mu_{moon}$=7.718260968E2 n.mi.$^3$/sec$^2$

## TK! Solver Model

Rule Sheet
S Rule
* Rap=a*(1+e)
* Rper=a*(1-e)
* Ra=a*(1-e^2)/(1+e*cos(nua))
* cos(acos(-1)-nua)=Rmoon/Ra
* TP=2*acos(-1)*sqrt(a^3/mu)
* tcomsm=(2*TP*(acos(-1)-nua)/(2*acos(-1)))/60

* Rb=a*(1-e^2)/(1+e*cos(nub))
* cos(nub)=Rmoon/Rb
* tcomse=(TP*(1-(nua-nub)/(2*acos(-1))))/60

* altap=Rap-Rmoon

Variable Sheet

| St Input | Name | Output | Unit | Comment |
|---|---|---|---|---|
| 938.49353 | Rmoon | | n.mi. | Radius of the Moon |
| 998.49353 | Rper | | n.mi. | Radius of ellipse at periapsis (60 n.mi. alt.) |
| | Rap | 998.49353 | n.mi. | Radius of ellipse at apoapsis |
| | Ra | 998.49353 | n.mi. | Radius of ellipse at point A |
| | Rb | 998.49353 | n.mi. | Radius of ellipse at point B |

| | | | | | |
|---|---|---|---|---|---|
| L | | altap | 60 | n.mi. | Altitude of the satellite at apoaposis |
| | | a | 998.49353 | n.mi. | Semi-major axis of ellipse |
| L | 0 | e | | unitless | Eccentricity of ellipse |
| | | nua | 2.7931614 | radians | Angle between periapsis and point A |
| | | nub | .34843126 | radians | Angle between periapsis and point B |
| | 771.8261 | mu | | n.mi.^3/s | Moon's gravitational parameter |
| | | TP | 7135.7236 | sec | Period of the elliptical orbit |
| L | | tcomsm | 13.19028 | min | Communication time possible between the satellite and Moon |
| L | | tcomse | 72.654643 | min | Communication time possible between the satellite and Earth |